# PERFORMANCE ANALYSIS OF CHEMICAL ROCKET MOTOR

By

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# PERFORMANCE ANALYSIS
# OF
# CHEMICAL ROCKET MOTOR

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*By*
KIRAN R. MAGIAWALA



to the

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AUGUST 1973

TO MY MOTHER

CERTIFICATE

Certified that the thesis "Performance Analysis of Chemical Rocket Motor" has been carried out under our supervision and it has not been submitted elsewhere for a degree.

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TABLE OF CONTENTS

PAGE

# LIST OF TABLES

LIST OF FIGURES

# LIST OF SYMBOLS

| | |
|---|---|
| $\dot{m}$ | Mass flow rate through nozzle. |
| $\dot{w}$ | Weight flow rate through nozzle. |
| $F$ | Thrust. |
| $I_{sp}$ | Specific impulse. |
| $C_F$ | Thrust Coefficient. |
| $C^*$ | Characteristic velocity. |
| $u_{eq}$ | Equivalent or effective exhaust velocity. |
| $v$ | Linear flow velocity. |
| $C_d$ | Coefficient of discharge. |
| $A_{in}$ | Area of injector hole. |
| $\rho$ | Density of fluid. |
| $\Delta p$ | Pressure drop across injector. |
| $W$ | Molecular weight. |
| $\epsilon$ | Exit area of nozzle/throat area of nozzle. |
| $n$ | Number of moles. |
| $p$ | Pressure in atmospheres. |
| $A$ | Equivalent formula weight. |
| $m$ | Number of moles of hydrogen. |
| $R$ | Equilibrium constant based on number of moles. |
| $\Delta Hf^o$ | Standard heat of formation at 298.16 °K |
| $H_o^T$ | Sum of sensible enthalpy and chemical energy at temperature T and standard condition. |
| $H_o^o$ | Chemical energy at 0 °K and standard condition. |
| $C_p^o$ | Molar specific heat at constant pressure and standard conditions. |

| | |
|---|---|
| $T$ | Temperature. |
| $\gamma$ | Ratio of specific heats. |
| $\alpha$ | Divergence angle of nozzle. |
| $g$ | Gravitational constant. |
| $J$ | Conversion factor from thermal to mechanical energy. |
| $S_T^o$ | Molar entropy at standard condition. |
| $\dot{R}$ | Universal gas constant. |
| $X$ | Mole fraction. |
| $D$ | Diameter. |
| $\bar{\ }$ | Average value as a function of temperature. |
| $\bar{\ }^*$ | Particular average value. |
| c | Combustion chamber. |
| t | Throat of the nozzle. |
| e | Exit of the nozzle. |
| i,j | $i^{th}$ or $j^{th}$ species. |

# SYNOPSIS

Experimental and theoretical investigations are made to study performance of a gaseous rocket motor.

Experimental work is carried out at different mass flow rates, at various oxidizer fuel ratios, using nozzles of varying expansion ratios. The flame-in limits are also established.

Theoretical investigations include (a) Determination of equilibrium temperature and composition of chemical reaction, (b) performance parameter calculations by frozen flow technique and (c) performance parameter calculations by equilibrium flow technique.

Computer programmes are developed for use on IBM 7044/1401 system. First programme takes care of part (a). It allows simultaneous high speed calculations of equilibrium composition and temperature for $H_2 - O_2 - N_2$ system, for combustion at constant pressure and adiabatic conditions. Second and third programme deals with part (b) and (c). They incorporate basic assumptions of analysis. Satisfactory results are obtained within maximum error limit of $\pm$ 3%.

## SECTION I

## EXPERIMENTAL INVESTIGATIONS OF STATIC PERFORMANCE OF A CHEMICAL ROCKET MOTOR

# CHAPTER I

## INTRODUCTION

1.1 General

1.2 Work Related to Performance Analysis

1.3 Present Work

## 1.1 General

A rocket engine is capable of operating independently of its environmental medium because it carries its own propellants, both fuel and oxidiser. According to the type of the energy source used, rocket engine can be classified as chemical rocket engines, nuclear rocket engines, electrical rocket engines and solar rocket engines.

In chemical rocket engines, the energy from a high pressure combustion reaction of fuel and oxidising chemicals (propellants) adds to the heating of reaction product gases to a very high temperature. The high temperature gases thus obtained are subsequently expanded in a nozzle which permits conversion of their heat energy into kinetic energy. Physical state of the propellants used, once again classifies the chemical rocket engines, viz. solid propellant rocket engines, gaseous propellant rocket engines and hybrid rocket engines. Controlability of liquid propellant rocket engines has made them more versatile. Though solid propellant rockets are used sometimes in missiles and RATO applications. Gaseous propellant rockets have found limited applications as attitude control systems in space vehicles. Chemical additives for electron supression in rocket flames have been studied by Farber & Srivastava (1, 2).

### 1.2 Work Related to Performance Analysis

A great deal of work was proposed and had been succesively carried out during and after 2nd world war.

Sutton (3) and Barrare (4) have discussed at length the subject of rocket testing and many other experimental techniques in their text books. Sutton (5) and Youngquist (6) also tried to represent the thermochemistry of rocket propellants and testing ofliquid propellant rocket motor as a subject. They carried out some experiments with gasoline - liquid oxygen rockets and nitric acid - aniline rockets and after determining theoretical performance of the same, a fruitful comparision was established.

Tenebaum (7) and Anderten (8) discussed at length about testing of the giant rocket engines. Geothert (9) and RMI (10) also discussed the testing of rocket motor under simulated conditions.

Mottram (11) made a systematic survey of measurements of performance of liquid rocket motors. He also reviewed the special requirements of all necessary instruments used for testing of rocket motors. Bierlin (12) summarised the methods of thrust measurements, both direct and indirect. Venn (13), Jones (14) and Atman (15) worked on instrumentations for rocket motor testing e.g. on various oscilloscopes, pressure recorders etc. Bowers (16) and Shafer (17) did remarkable work in flow measurement and calibration of flow measuring instruments. Exhaust gas analysis was done by Bear (18).

## 1.3 Present Work

The present work includes the study of static performance of $H_2 - O_2$ gaseous rocket motor unit. Following major experiments are performed :

(1) Injector flow analysis

(2) Flame-in limits establishment

(3) Thrust meter analysis

(4) Determination of performance parameters at different mass flow rates at various oxidizer fuel ratios using nozzles of varying expansion ratios.

# CHAPTER II

# EXPERIMENTAL

2.1 DESCRIPTION OF APPARATUS

- 2.1.1 Thrust Chamber Assembly
- 2.1.2 Propellant Feed System
- 2.1.3 Valves and Controls
- 2.1.4 Other Components and Mounts

2.2 FABRICATION

2.3 EXPERIMENTAL PROCEDURE

- 2.3.1 Injector Flow Analysis
- 2.3.2 Flame-in Limits Establishment
- 2.3.3 Thrust Meter Analysis
- 2.3.4 Determination of Performance Parameters

## 2.1 Description of Apparatus

All the experiments are performed on a LABROC VI assembly provided by Astrosystem International Inc. U.S.A. Front and rear view of the unit are shown in Fig. (1) and Fig. (2). This rocket motor operates on hydrogen oxygen (commercial gas) propellants. The assembly can be examined in detail with following subdivisions.

### 2.1.1 Thrust Chamber Assembly

The entire unit (see Fig. 3) is made up of high temperature nickle alloy. It consists of injector (Fig. 4), combustion chamber (Fig. 5) and exhaust nozzle (Fig. 6).

Injector is of direct impingment type. Two jets of hydrogen gas are made to impinge central horizontal jet of oxygen gas at an angle of approximately 20°. Prior to as well as after completion of every firing, purging of combustion chamber is done with nitrogen gas. The combustion chamber is cylindrical, with provision for oxygen inlet, nitrogen inlet, hydrogen inlet, thrust knob, water inlet and outlet connections and nozzle mounting arrangements. Nozzles of various expansion ratios can be attached to the combustion chamber with metal 'O' ring seal. The entire assembly is water cooled.

### 2.1.2 Propellant Feed System

Commercial gas cylinders for hydrogen, oxygen, and nitrogen serves the purpose of propellant storage tanks. All these gases are made available to the combustion chamber via regulator.

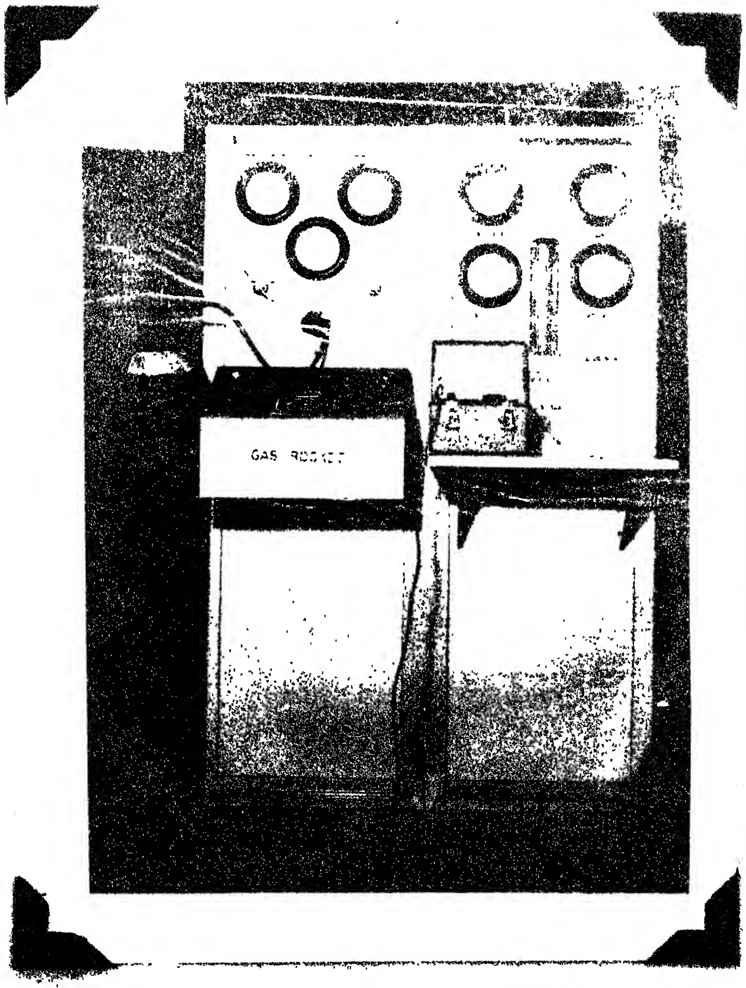

FIG.1_FRONT VIEW OF LABROC VI

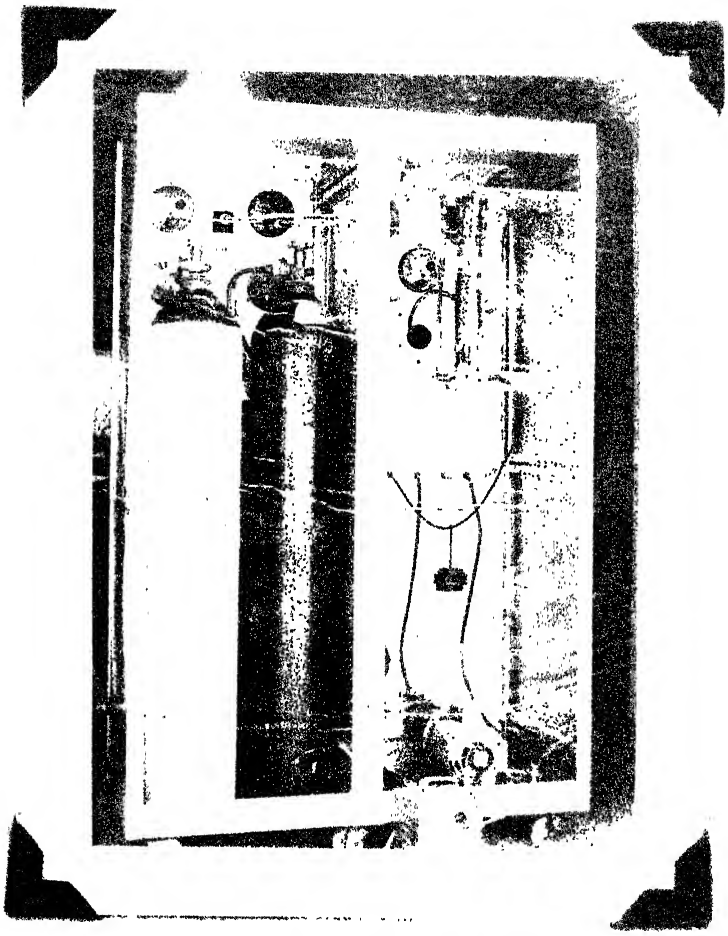

FIG.2_REAR VIEW OF LABROC VI

FIG. 4_ IMPINGING STREAM TYPE, TRIPLE HOLE INJECTOR

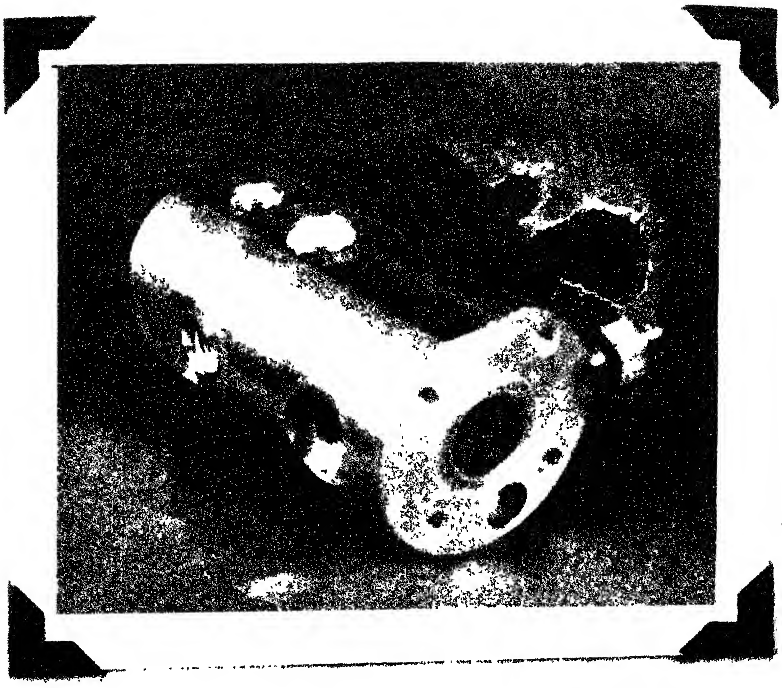

FIG. 5_WATER-COOLED COMBUSTION CHAMBER

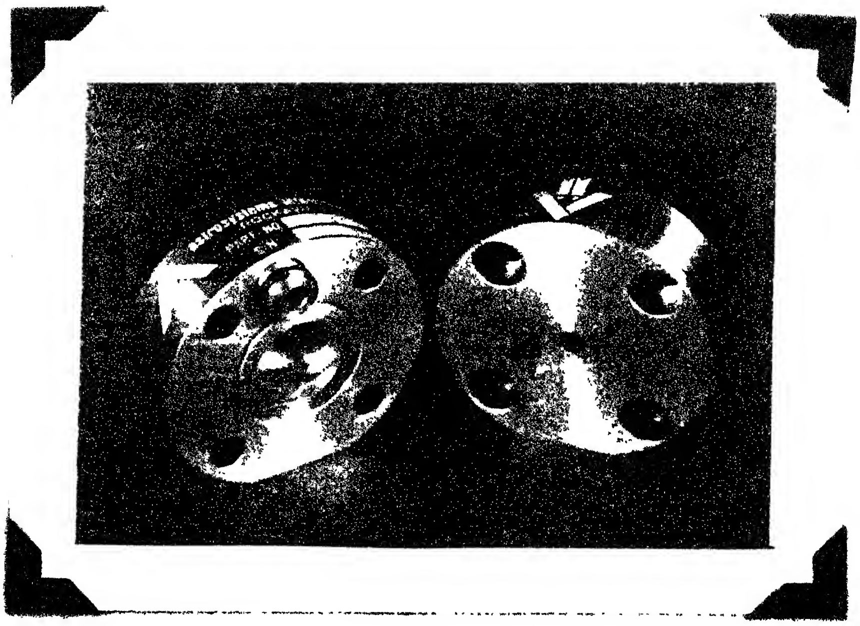

FIG.6_ROCKET EXHAUST NOZZLES

This regulates both pressure and mass flow rate of the gases. After regulaters there follows the solonoid operated valves. For details see Fig. (7).

2.1.3 Valves and Controls

Control circuit diagram is shown in Fig. (8).

Solenoid operated valves and other check valves are so connected in control circuit that, when water level is optimum which puts water switch on, both oxidiser and fuel solenoid valves are on open position, with fire switch on. When shut down switch is on only nitrogen gas will purge through gas circuits. Check switches and indicator bulbs are also provided on the console to check the master power, water level sufficiency, spark check and running time meter operation. Water switch is adjusted to minimum level of 7 lb/min. flow while running time meter is set at minimum of 75 Psig. Relays and timers are also used in the circuit which allow for a purging time of 17 seconds for nitrogen gas.

2.1.4 Other Components and Mounts

This includes cooling water circulation pump, rotometer for flow measurement, thermometers for water inlet and outlet temperature to the thrust chamber, chamber pressure measurement gauge, propellant storage tank pressure gauge, thrust meter.

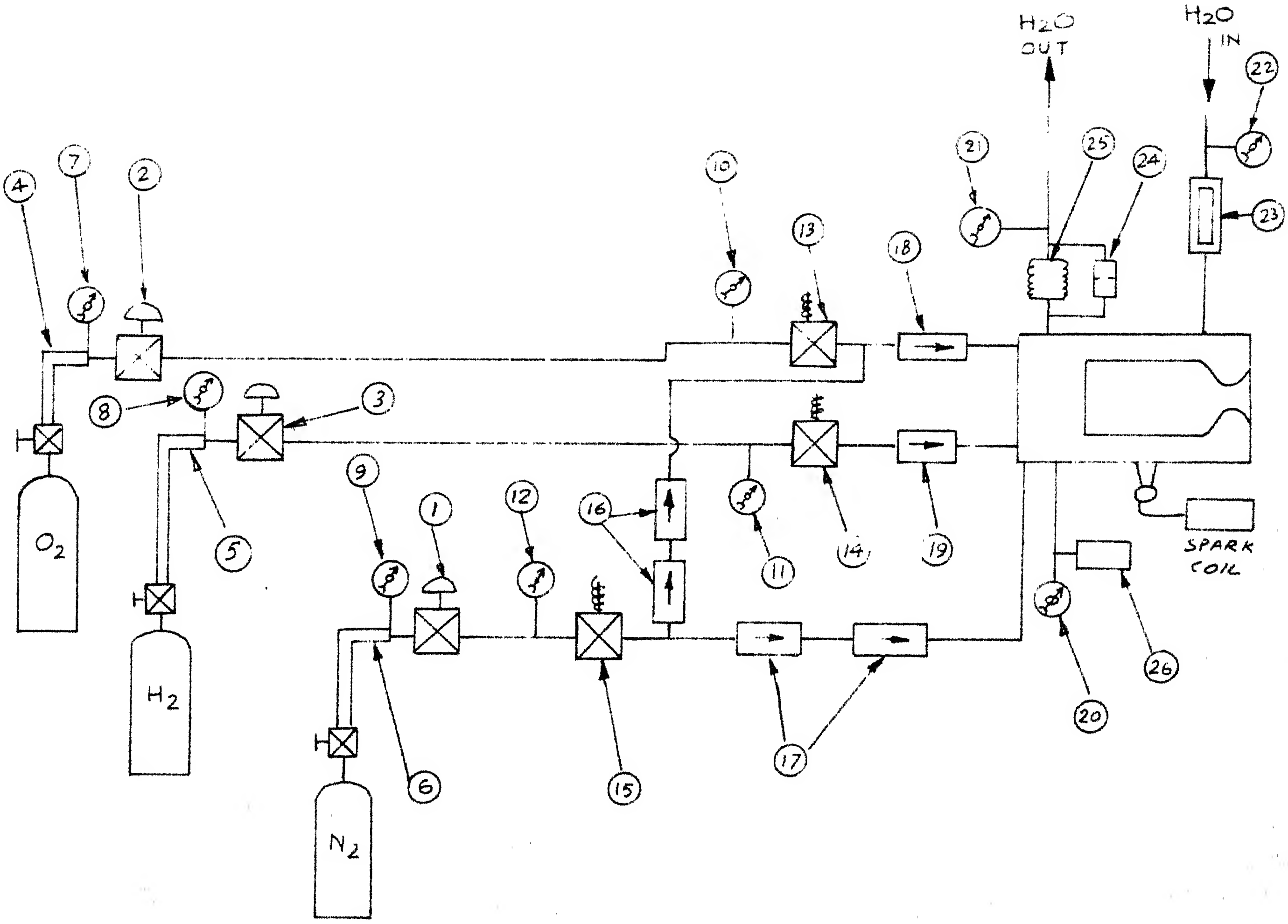


FIG. 7_ PROPELLANT FEED SYSTEM FOR LABROC VI

Nomenclature For Figure (7)

| | |
|---|---|
| 1 | Pressure regulator nitrogen purge |
| 2 | Pressure regulator oxygen supply |
| 3 | Pressure regulator hydrogen supply |
| 4 | High pressure line oxygen supply |
| 5 | High pressure line hydrogen supply |
| 6 | High pressure line nitrogen supply |
| 7 | Pressure gauge oxygen tank |
| 8 | Pressure gauge hydrogen tank |
| 9 | Pressure gauge nitrogen tank |
| 10 | Pressure gauge oxygen supply |
| 11 | Pressure gauge hydrogen supply |
| 12 | Pressure gauge nitrogen supply |
| 13 | Solenoid valve oxygen propellant valve |
| 14 | Solenoid valve hydrogen propellant valve |
| 15 | Solenoid valve nitrogen purge |
| 16 | Check valves nitrogen purge, oxygen system |
| 17 | Check valves nitrogen purge, hydrogen system |
| 18 | Check valve oxygen system |
| 19 | Check valve hydrogen system |
| 20 | Pressure gauge chamber pressure |
| 21 | Thermometer cooling water outlet temperature |
| 22 | Thermometer cooling water inlet temperature |
| 23 | Flow meter, cooling water |
| 24 | Orifice, cooling water flow orifice |
| 25 | Pressure switch, cooling water |
| 26 | Pressure switch, chamber pressure |

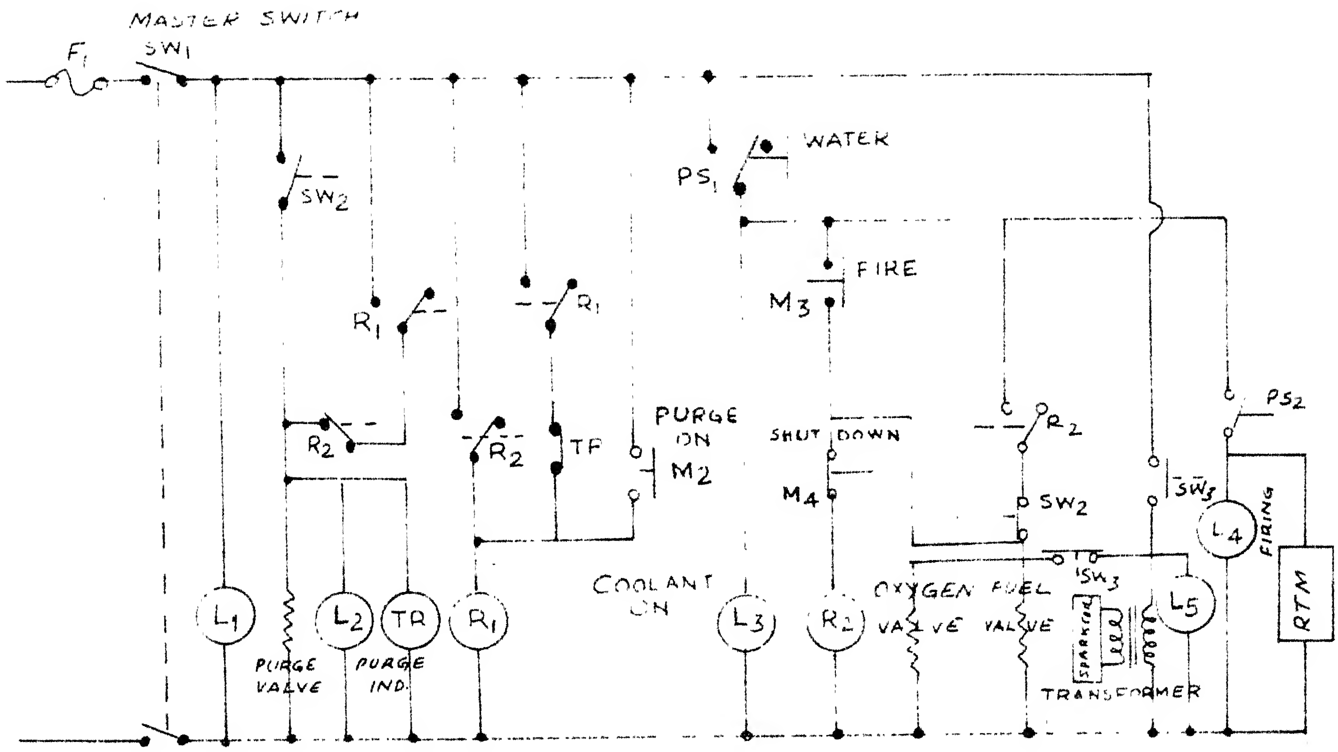


FIG. 8_ELECTRICAL CONTROL SYSTEM FOR LABROC VI

Nomenclature for Figure (8)

| | | |
|---|---|---|
| $F_1$ | Fuse | |
| $L_1$ | Pilot light, Power (white) | |
| $L_2$ | Pilot light, Purge (green) | |
| $L_3$ | Pilot light, Coolant (blue) | |
| $L_4$ | Pilot light, firing (red) | |
| $L_5$ | Pilot light, spark check (amber) | |
| $M_2$ | Purge off momentary contact button | SPST |
| $M_3$ | Fire momentary contact button | SPST |
| $M_4$ | Shutdown momentary contact button | SPST |
| $PS_1$ | Coolant pressure switch | |
| $PS_2$ | Chamber pressure switch | |
| $R_1$ | Purge relay | DPST |
| $R_2$ | Fire relay | TPDT |
| $SW_1$ | Master switch | DPST |
| $SW_2$ | Emergency shut down switch | DPDT |
| $SW_3$ | Circuit spark check switch | |
| Transformer | | |
| RTM | Running time meter | |
| Spark Coil | | |
| TR | Purge timer | |

## 2.2 Fabrication

This includes fabrication and calibration setup of thrust meter. The thrust meter was fabricated to measure the thrust of the rocket motor. See Fig. (9).

Bounded wire type of resistance strain gauge was used and it was mounted on thin steel plate of 'I' shape. The thrust knob from the rocket motor was made to touch the other side of 'I' shaped plate with the help of initial load see Fig. (10).

Output of the strain gauge was taken to strain gauge indicator. Proper bridge connections are essential. Calibration of this thrust meter was done before each set of firing, with the help of known weights.

## 2.3 Experimental Procedure

Prior to actual test run calibration of injector for gas flow is done together with thrust meter calibration with known weights. Experiments of establishment of flame-in limits and performance parameter measurements were also done. The details are given below.

### 2.3.1 Injector Flow Analysis

LABROC VI operating manual provides the injector flow calibration curve for oxygen-methane propellant system. As shown in Appendix A.1, these can be deduced for oxygen - hydrogen propellant system.

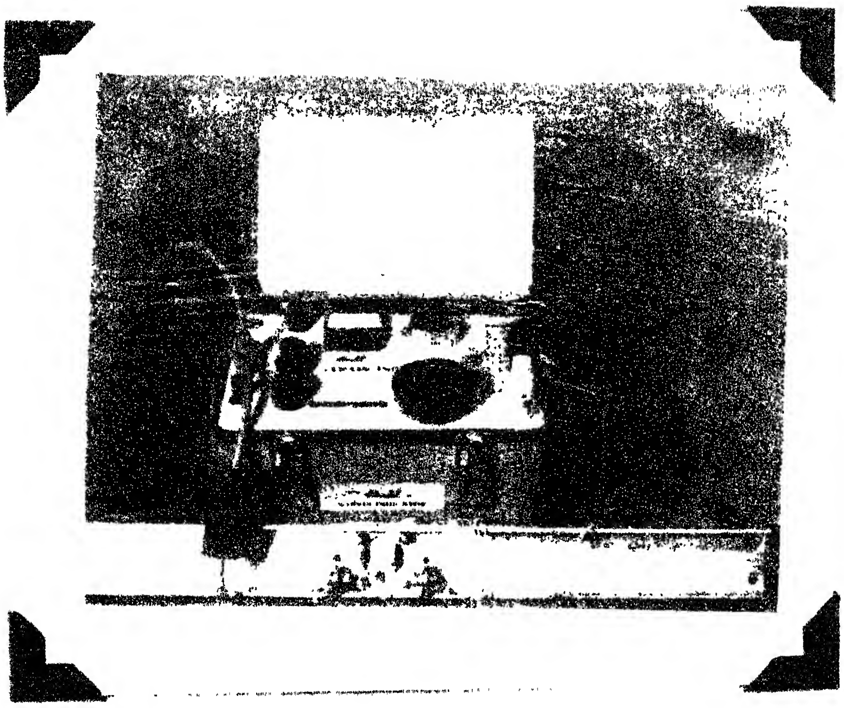

FIG.9_ THRUST MEASURING UNIT

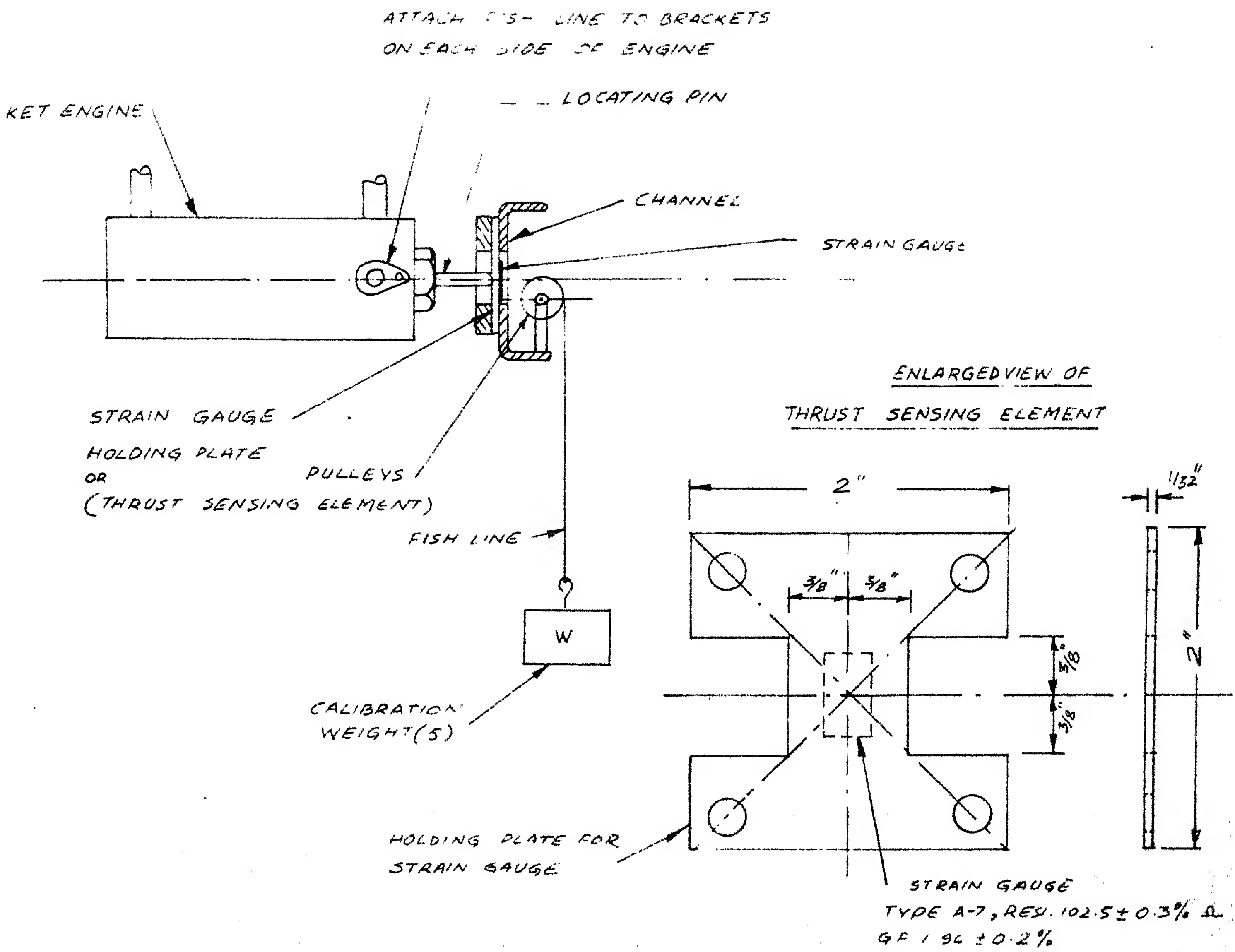


FIG. 10 _ ENGINE AND THRUST MOUNT ASSEMBLY

2.3.2 Flame-in Limits Establishment

Here for various fixed settings of either fuel or oxidiser, the setting of the other one is so varied until the flame just sets in. Combustion chamber pressure is measured at this instant. From regulator settings and injector calibration curves for hydrogen oxygen combination, O/F ratio is calculated. Fixed oxygen setting gives high O/F ratio flame-in limits while fixed hydrogen setting gives low O/F ratio flame-in limits. This is done so as to obtain readings with maximum control parameter provided with LABROC VI and to avoid dangerous explosions due to hydrogen leakage. Appendix A.2 deals with data, sample calculations and calculated readings.

2.3.3 Thrust Meter Analysis

Here, after the thrust knob is made to touch strain gauge plate with initial load, null point is adjusted on strain gauge indicator. Strain readings are taken for each incremental load. Appendix A.3 deals with data, sample calculations and calculated readings.

2.3.4 Determination of Performance Parameters

For a particular mass flow rate through a nozzle, various regular readings for hydrogen and oxygen are fixed up for different O/F ratios. The thrust meter is then calibrated as indicated in 2.3.3. After the engine purge, firing is made

according to instruction manual of LABROC VI. Readings of thrust meter, combustion chamber pressure, coolant flow rate and inlet and outlet temperature, etc. are taken at fixed O/F ratio, for a given mass flow rate. The similar readings are taken at different O/F ratios at the same mass flow rate and at different mass flow rates. The same is repeated with different nozzles. Appendix A.4 deals with data, sample calculations and calculated readings. Tables (3), (4), (5) and (6) are used to determine the necessary Panel settings.

# CHAPTER III

## RESULTS AND DISCUSSION

3.1 Injector Flow Analysis

3.2 Flame-in Limits Establishment

3.3 Thrust Meter Analysis

3.4 Performance Parameter Analysis

### 3.1 Injector Flow Analysis

A plot of the data, given in Table (1), Appendix A.1 is shown in Fig. (11). This figure is obtained from the operational manual of LABROC VI. Also a plot of the data given in Table (2), Appendix A.1 is shown in the Fig. (12).

From the Fig. (11) and Fig. (12) it can be seen that when Panel pressure in Psig vs. flow rate in lb/sec. is plotted on LOG x LOG scale, their variations are linear. This is in good agreement with the equation (A.1.1) considered on LOG x LOG scale for incompressible flow

$$\dot{m} = C_d \, A_{in} \sqrt{2 \rho \Delta p}$$

$$\text{or} \qquad \dot{m} = \text{Const} \sqrt{\Delta p}$$

Taking logerithm on both sides we have

$$\therefore \log (\dot{m}) = \log \text{const} + 1/2 \log \Delta p$$

for $\Delta p \sim p$ Panel this can be written as

$$\therefore \log (\dot{m}) = \log \text{ const} + 1/2 \log ( p \text{ Panel})$$

Also for methane, molecular weight is 16 gms./moles while for hydrogen it is only 2 gms./moles. Therefore for the same panel setting less amount of hydrogen gas is obtained due to difference in molecular weights and hence densities, for $C_d$ and $A_{in}$ remaining the same.

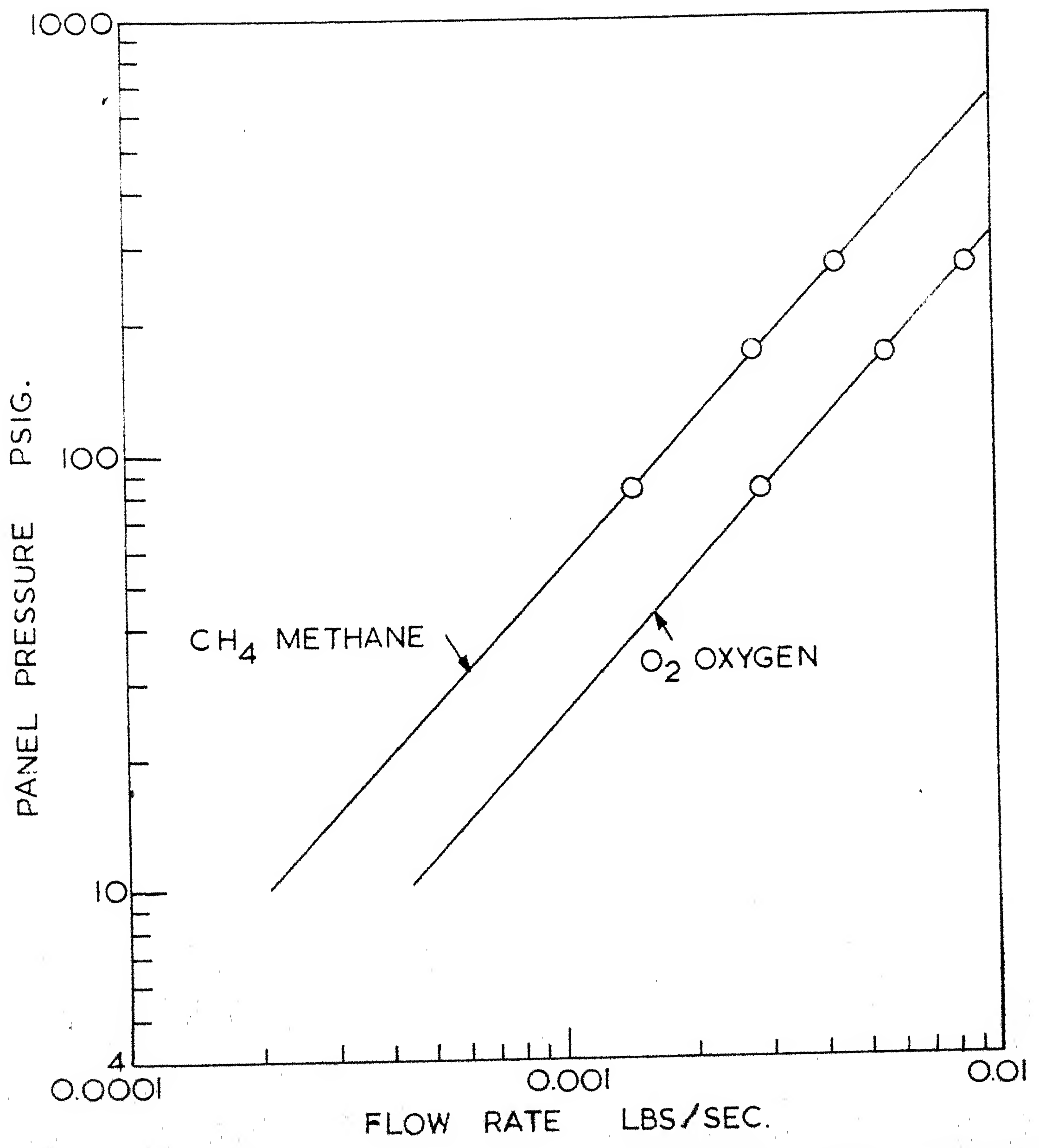


FIG. 11 _ INJECTOR FLOW CALIBRATION FOR GASEOUS PROPELLANTS METHANE AND OXYGEN

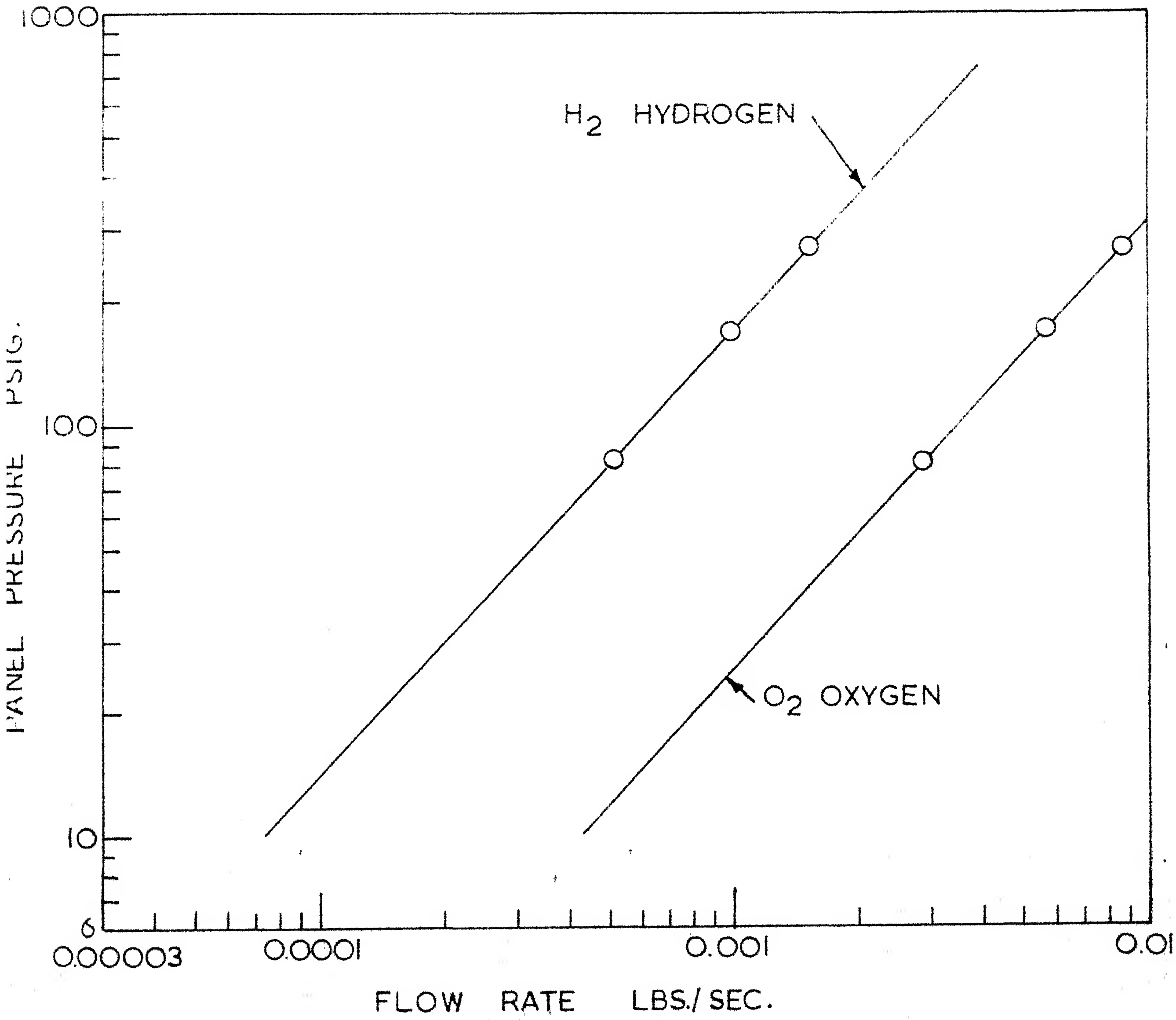


FIG.12 _ INJECTOR FLOW CALIBRATION FOR GASEOUS PROPELLANTS HYDROGEN AND OXYGEN

### 3.2 Flame-in Limits Establishment

A plot of the data given in the Tables (9) and (10) in Appendix A.2 for a particular rocket motor, nozzle combination is given in Fig. (13) using $H_2$ - $O_2$ fuels and oxidiser. Also a plot of the data given in the Tables (13) & (14) in Appendix A.2 for other rocket motor, nozzle combination is given in Fig. (14)

From the Fig. (13) and Fig. (14) it can be seen that a plot of combustion chamber pressure vs. O/F ratio establishes a zone for given rocket motor and nozzle combination. That is to say that the rocket motor must be operated within that zone to obtain the flame. It is also clear that this operating zone is quite wider for rocket motor and nozzle $\epsilon$ 1 : 1 than for rocket motor and nozzle $\epsilon$ 2.09 : 1.

### 3.3 Thrust Meter Analysis

The plots of datas given in Tables (15), (16), (17), (18), (19) and (20) in Appendix A.3 are given separately in Figs. (15), (16), (17), (18), (19) and (20) respectively.

From the Figures (15), (16), (17), (18), (19) and (20) it is clear that strain developed in the load cell designed is linearly proportional to the load applied. This also indicates that all the time, we are working within elastic limit of the gauge. For the same initial load of 2.11 lbs., for different sets, the initial and therefore the rest of the readings

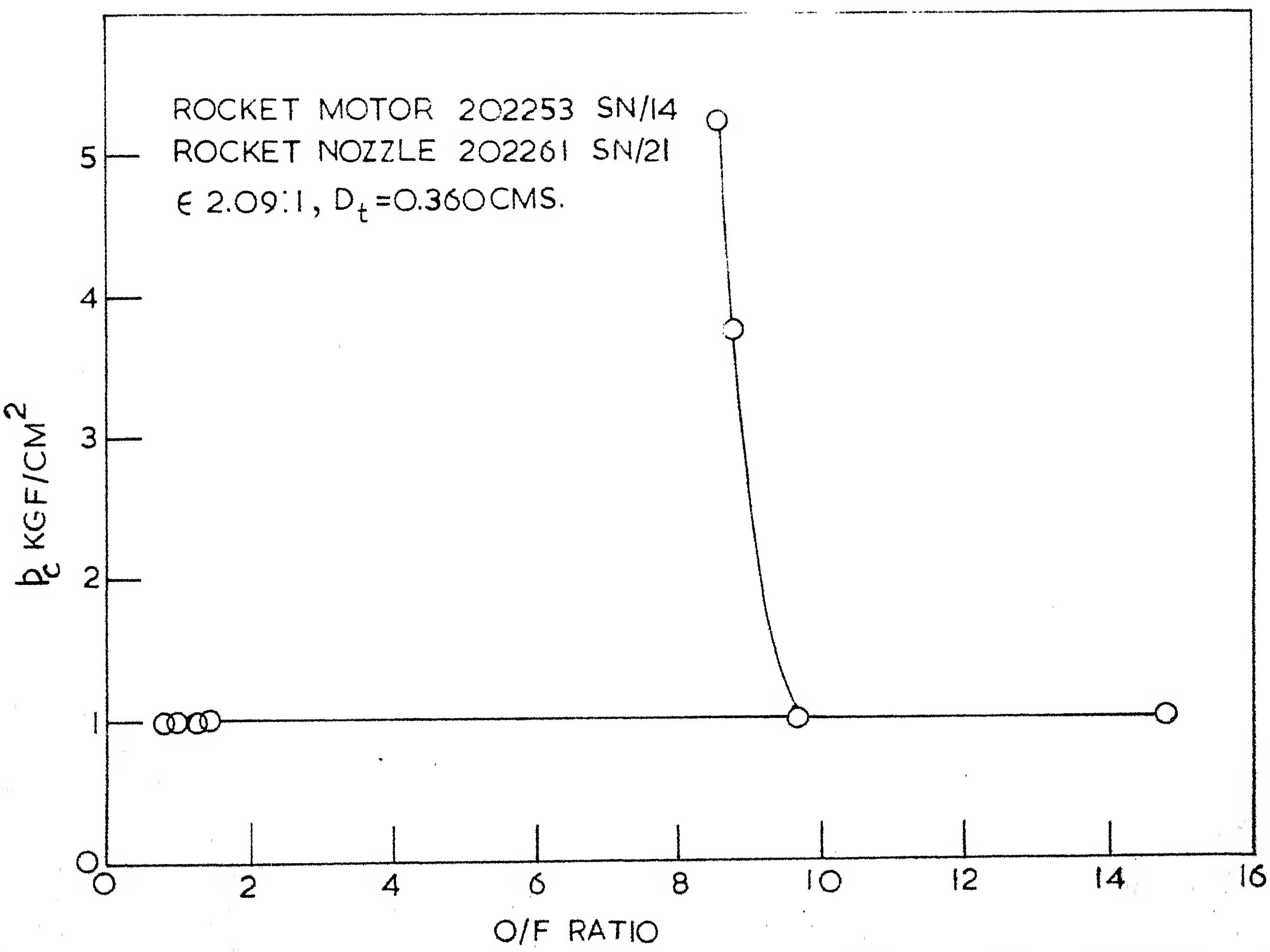


FIG.14_FLAME-IN LIMITS OF GASEOUS PROPELLANT ROCKET MOTOR USING HYDROGEN AND OXYGEN

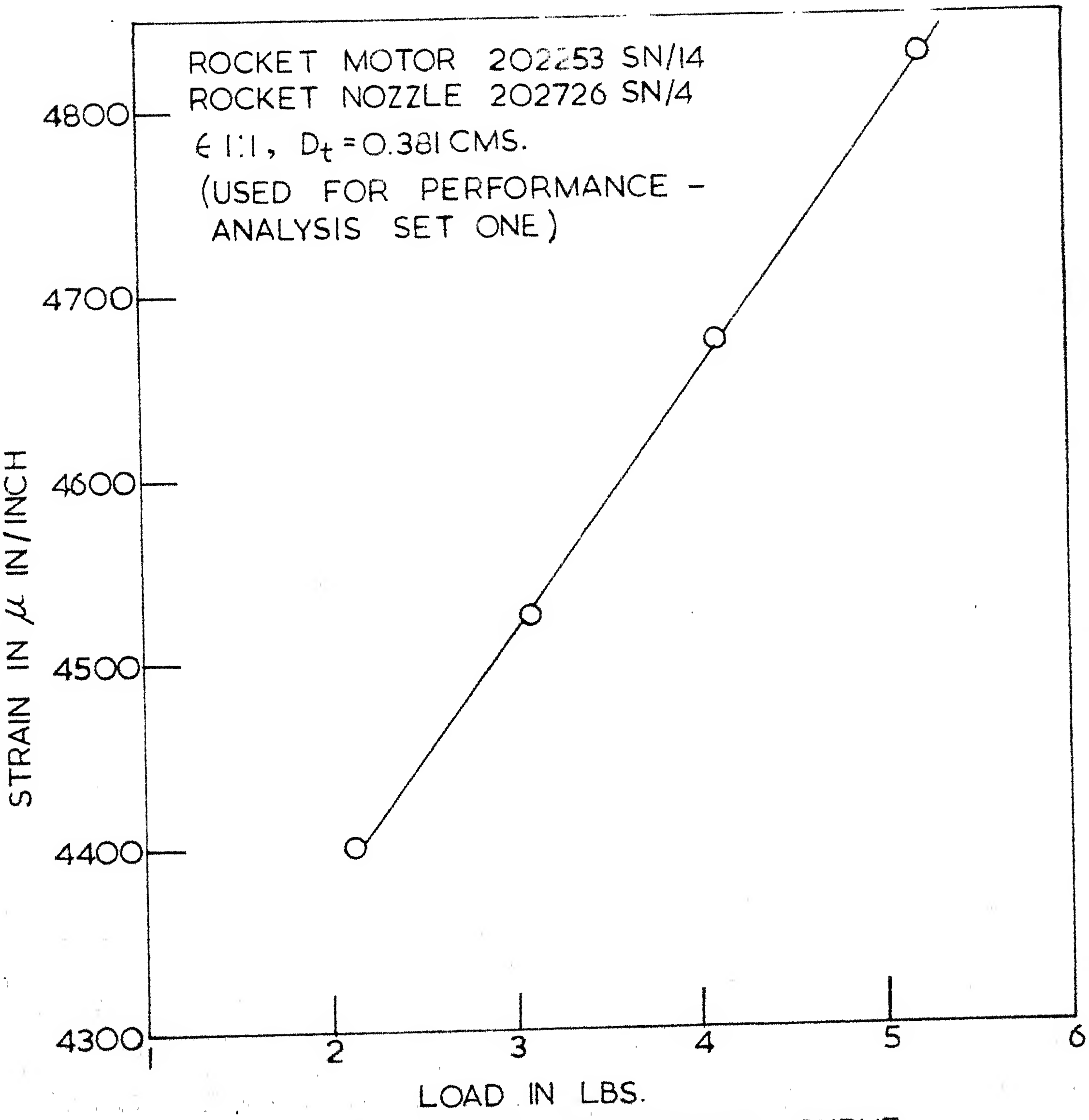


FIG.15_THRUST METER CALIBRATION CURVE

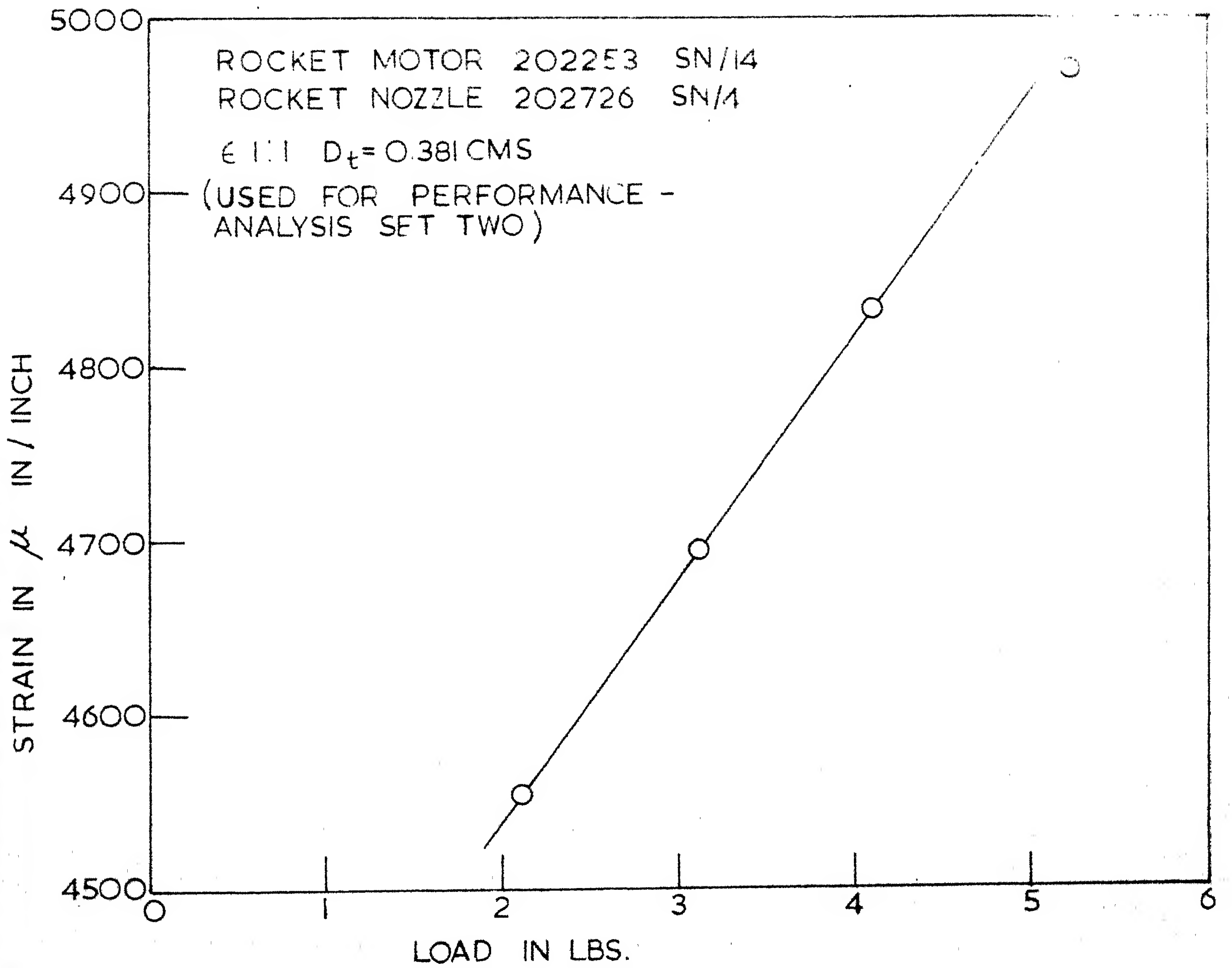


FIG 16_ THRUST METER CALIBRATION CURVE

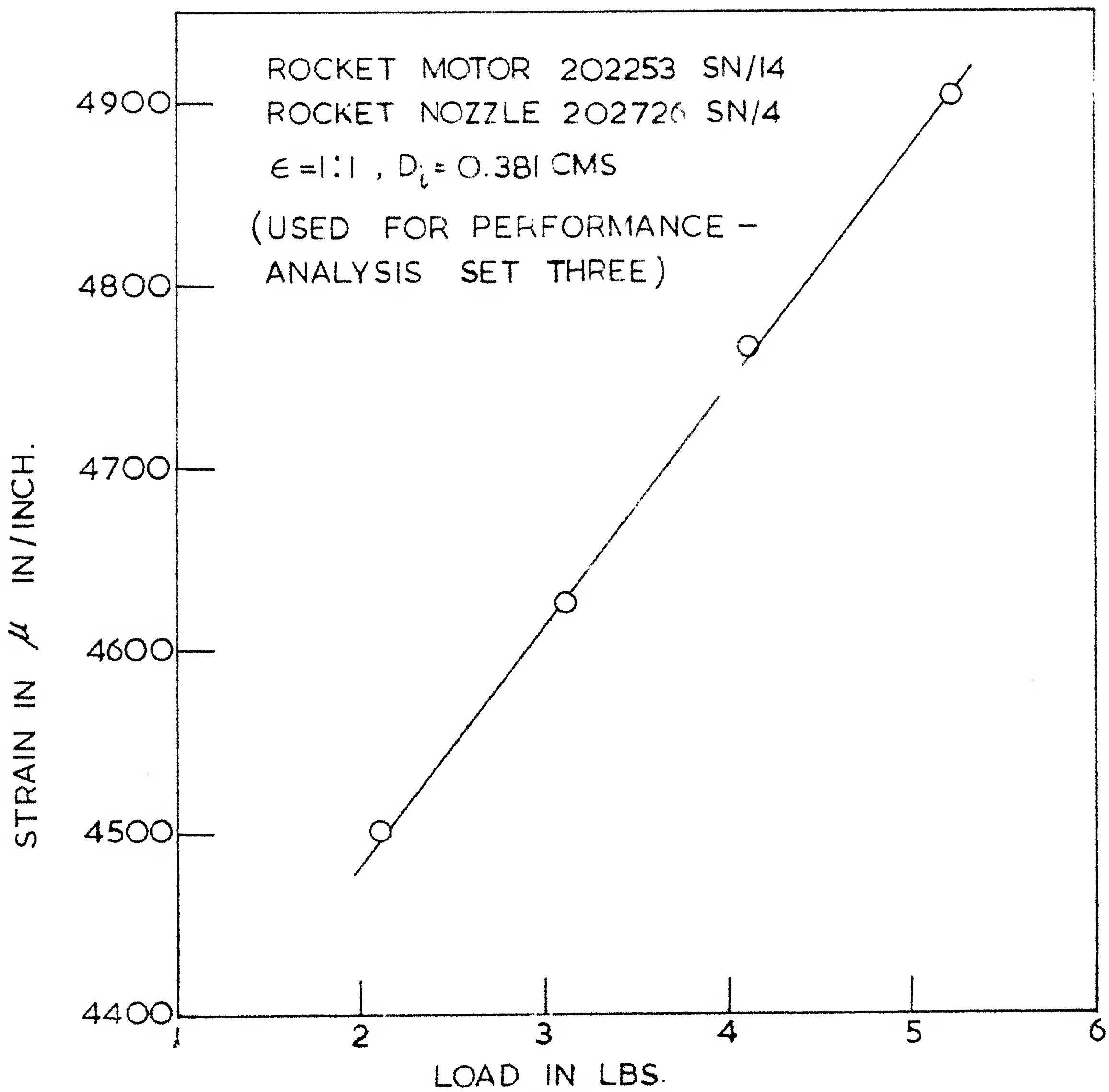


FIG.17_ THRUST METER CALIBRATION CURVE

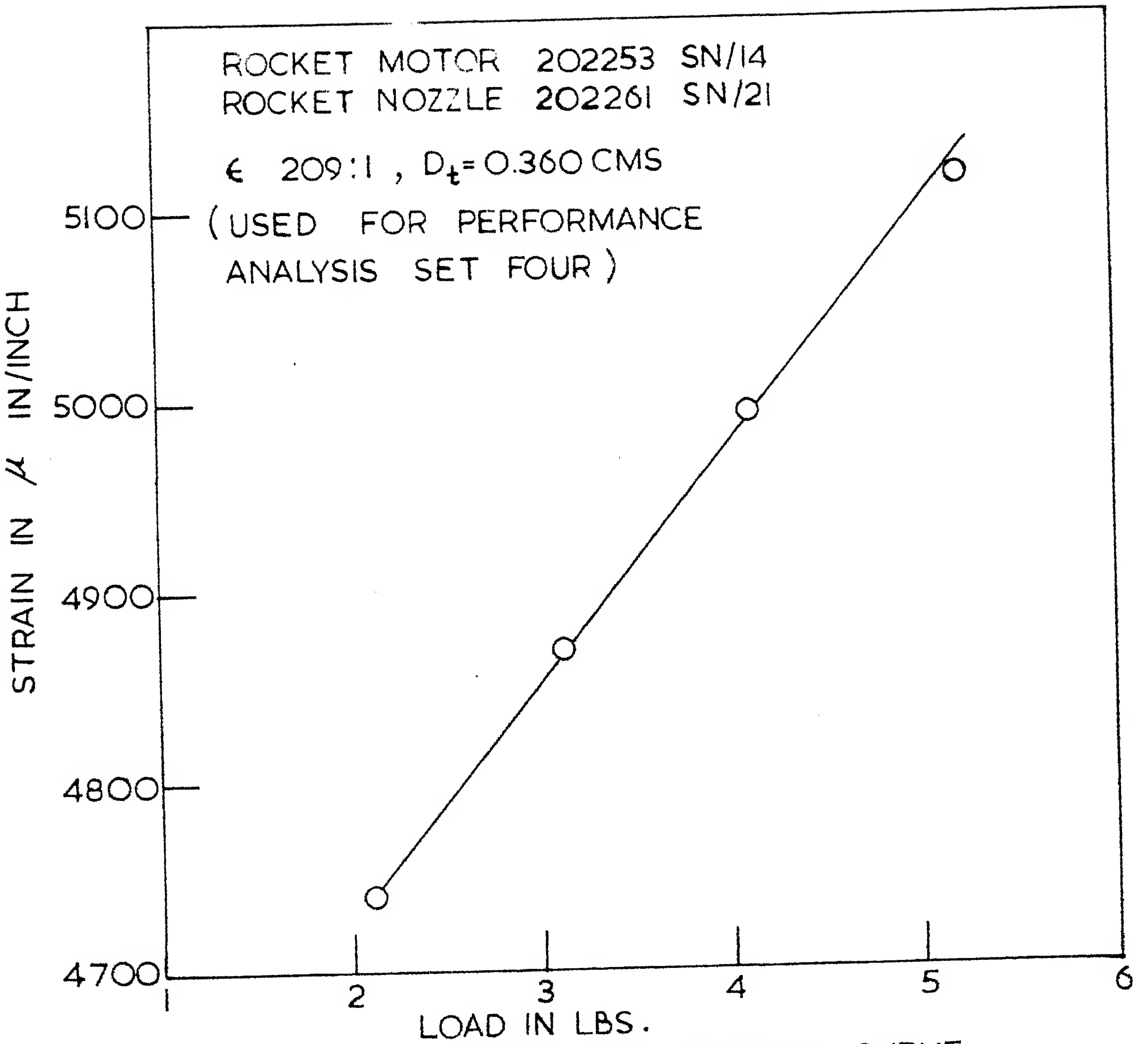


FIG.18 _ THRUST METER CALIBRATION CURVE

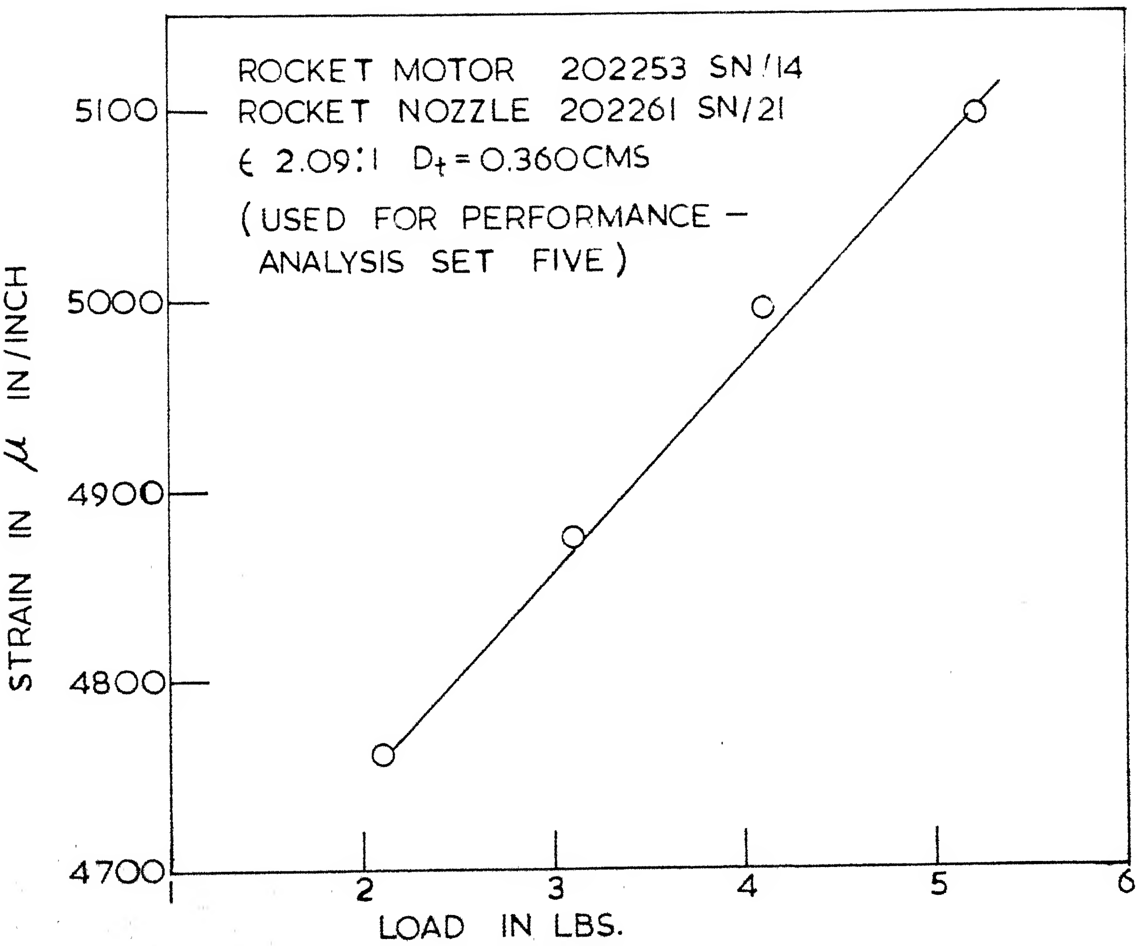


FIG.19_ THRUST METER CALIBRATION CURVE

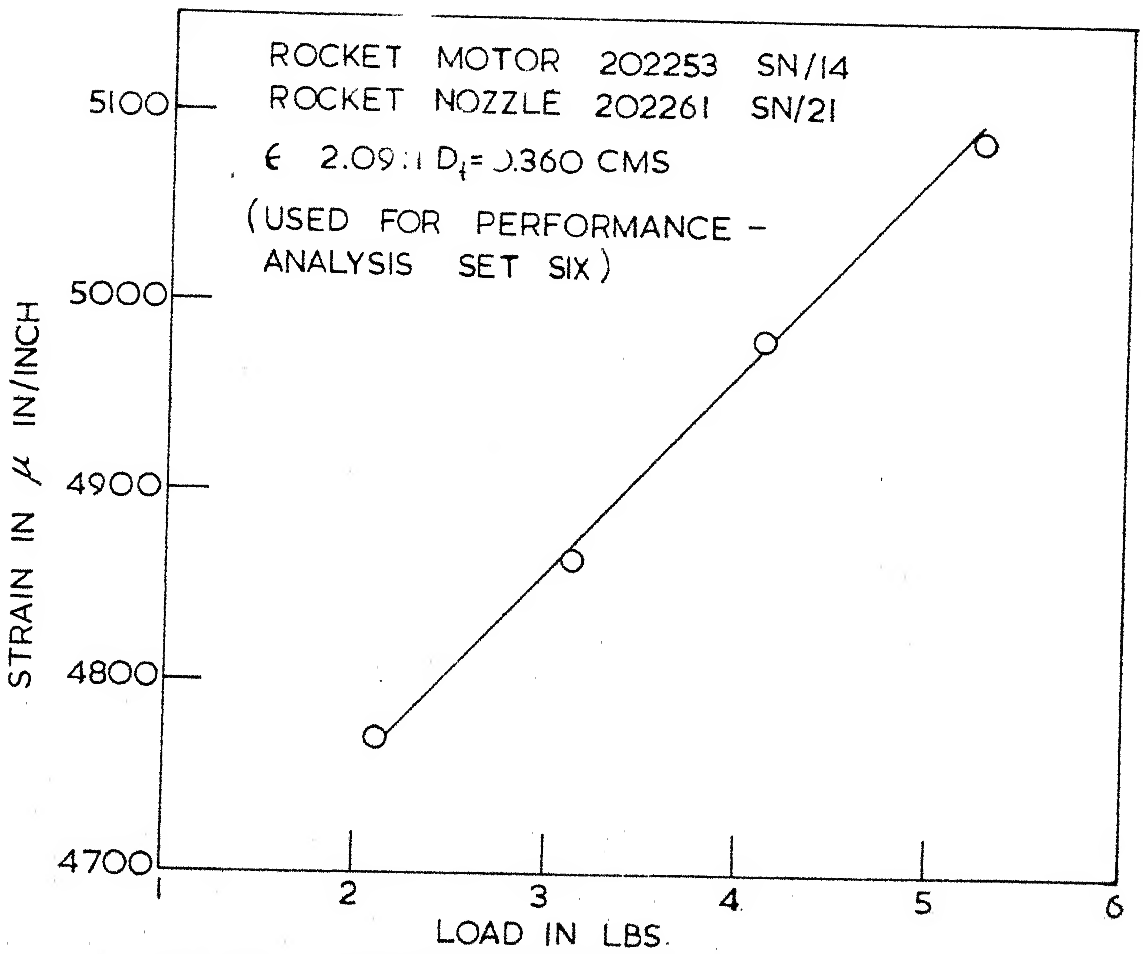


FIG. 20_ THRUST METER CALIBRATION CURVE

are slight different. This is probably due to the following reasons :

(1) Unequal temperature conditions

(2) unequal tensions in the strings

(3) Accidental blasts which happened on account of leakage of hydrogen gas from the solenoid operated valve and

(4) Velocity of air which may shift the readings due to oscillations of hanging weights.

3.4 Performance Parameter Analysis

The plots of the data given in Tables (22), (24), (26), (28), (30) and (32) in Appendix A.4 are given in the Figs. (21), (22), (23), (24), (25) and (26) respectively. These figures include the variation of performance parameters of gaseous propellant rocket motor using hydrogen and oxygen for O/F ratio varying from 4 to 8 and for mass flow rates varying from 0.009 lb/sec. to 0.013 lb/sec.

From the Figs. (21), (22), (23), (24), (25) and (26) for a particular set the following can be concluded .

When O/F ratio is decreasing

(1) Combustion chamber pressure increases

(2) Thrust and hence specific impulse also increases

(3) Characteristic velocity, which is the measure of combus-

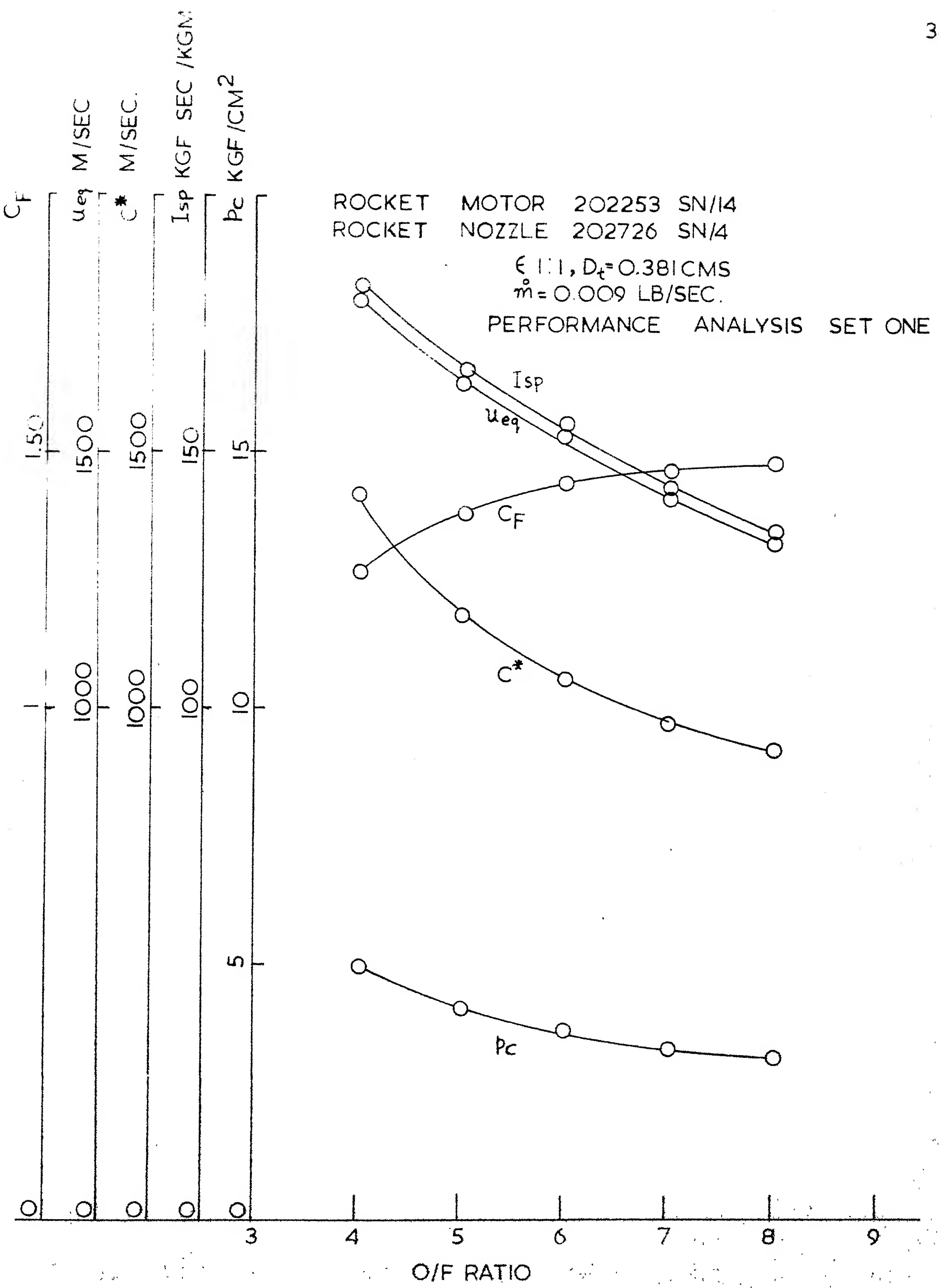


FIG. 21 _ VARIATION OF PERFORMANCE PARAMETERS OF A GASEOUS ROCKET MOTOR USING HYDROGEN AND OXYGEN, WITH O/F RATIO

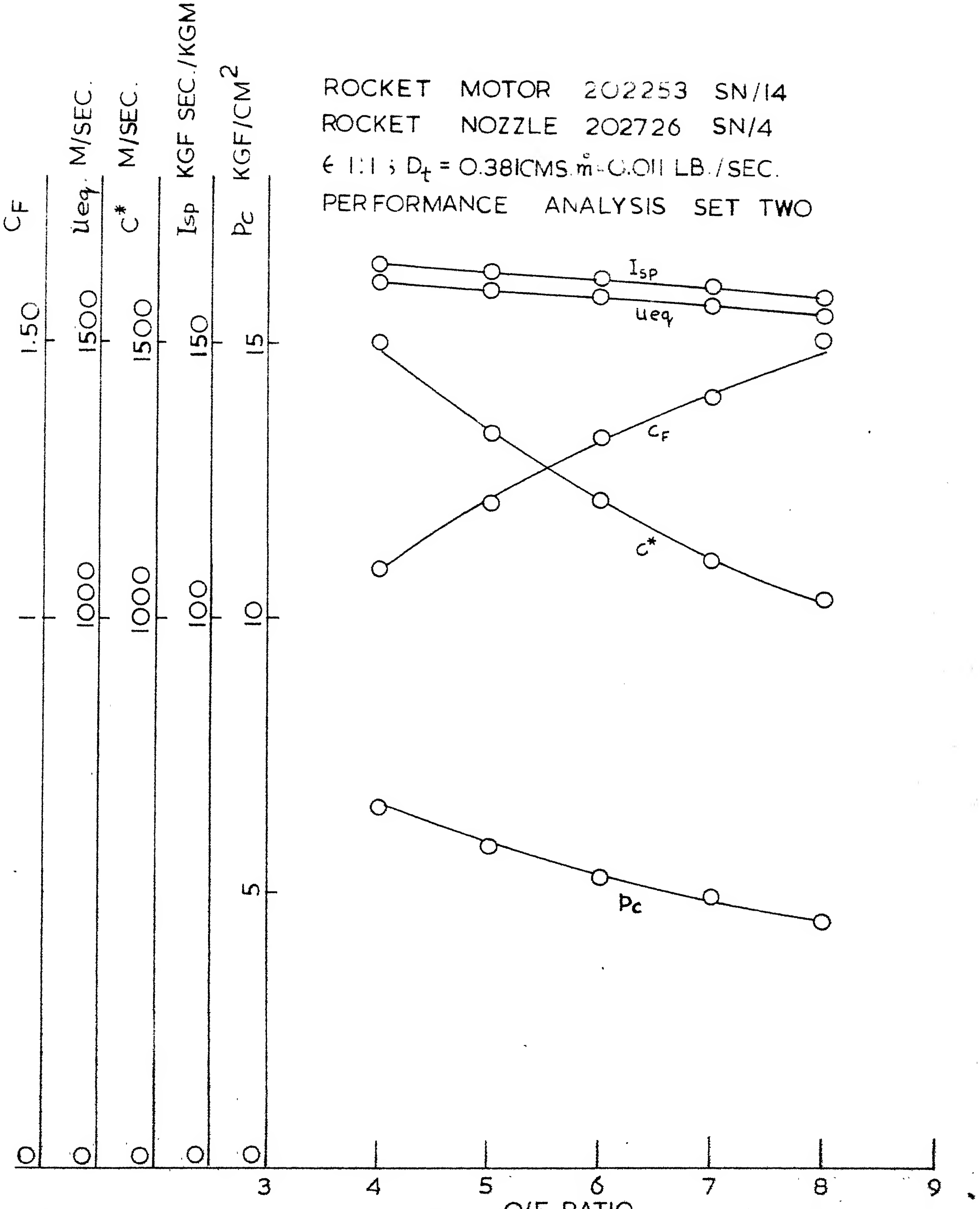


FIG. 22_ VARIATION OF PERFORMANCE PARAMETERS OF A GASEOUS ROCKET MOTOR USING HYDROGEN AND OXYGEN, WITH O/F RATIO

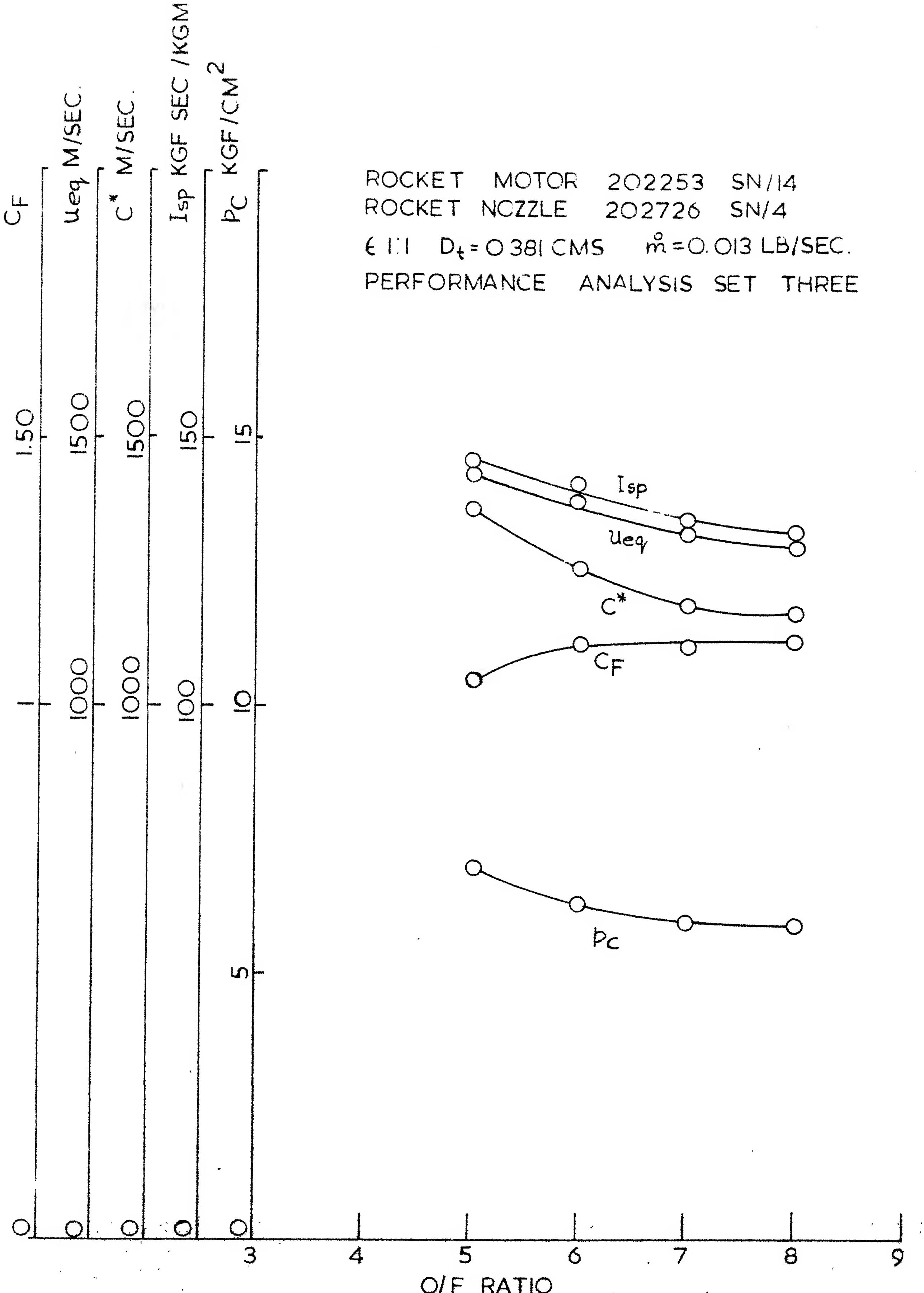


FIG. 23_VARIATION OF PERFORMANCE PARAMETERS OF A GASEOUS ROCKET MOTOR USING HYDROGEN AND OXYGEN, WITH O/F RATIO

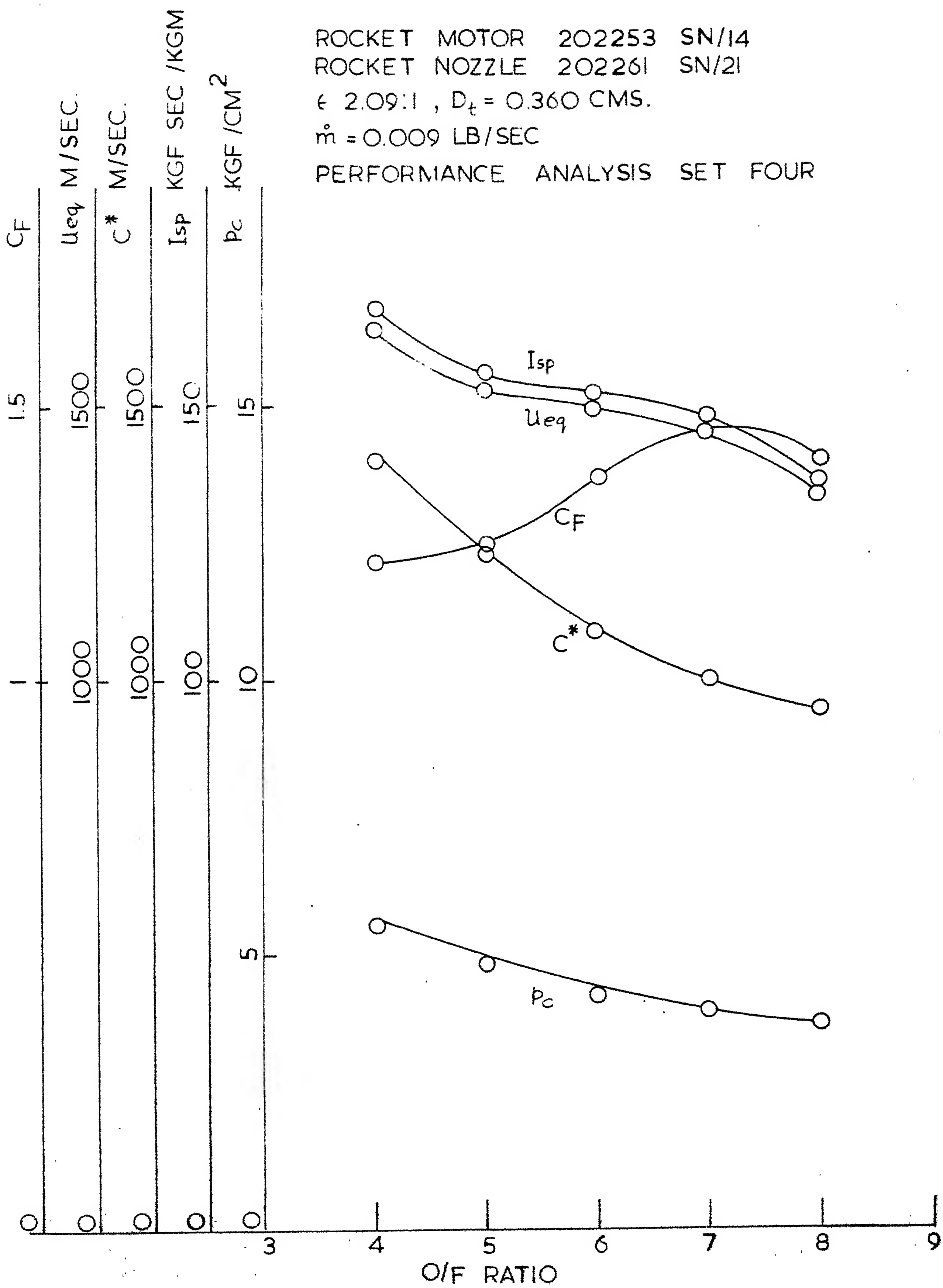


FIG. 24 _ VARIATION OF PERFORMANCE PARAMETERS OF A GASEOUS ROCKET MOTOR USING HYDROGEN AND OXYGEN, WITH O/F RATIO

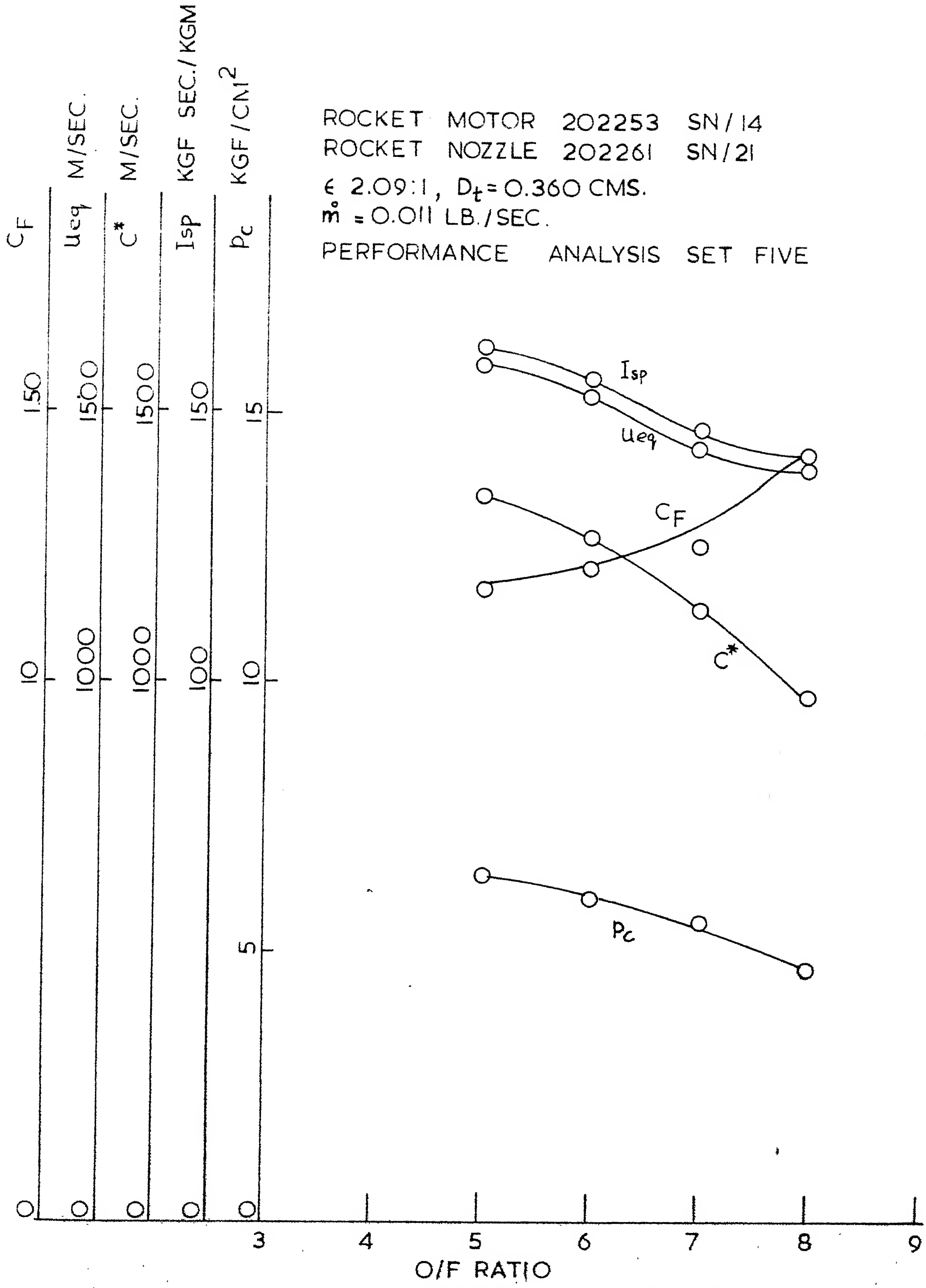


FIG. 25 VARIATION OF PERFORMANCE PARAMETERS OF A GASEOUS ROCKET MOTOR USING HYDROGEN AND OXYGEN, WITH O/F RATIO

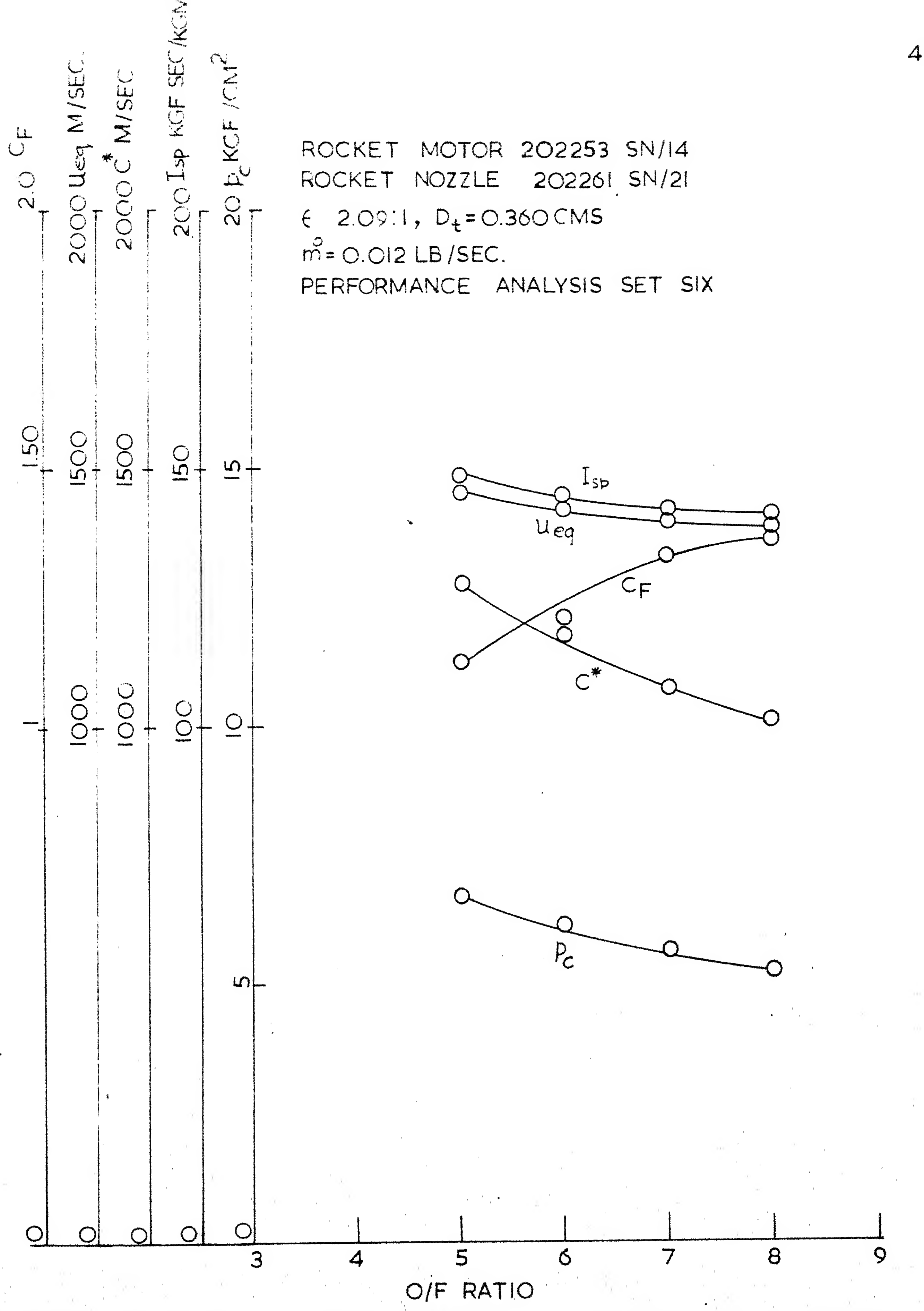


FIG. 26 _ VARIATION OF PERFORMANCE PARAMETERS OF A GASEOUS ROCKET MOTOR USING HYDROGEN AND OXYGEN, WITH O/F RATIO.

(4) Equivalent velocity also increases.

(5) Thrust coefficient which is the measure of nozzle working, reduces. The reason being the amount by which $p_c$ builds up being more than build up amount of F.

Conclusions (1) to (4) can be explained as under. Lower O/F ratio implies the higher settings of panel pressures. This leads to higher mixing pressure of hydrogen oxygen jet before ignition. On firing obviously this leads to higher pressure which indicates the availability of more amount of energy for subsequent conversion.

Now we will observe in detail Tables (22), (24) and (26). It is clear that for increase in mass flow rate thrust increases (exception set 3 for its first two readings). But the amount by which thrust increases is less than that of mass flow rate increase, resulting in the reduction of the specific impulse (observe last two readings of set 2 and for whole of set 3). Thrust coefficient variation range is largest for set 2 then comes for set 1 and set 3. Thus a kind of saturation occurs with set 3. Characteristic velocity for set 2 is larger than for set 1. Same is the case with them with equivalent velocity except last two readings. Equivalent velocity for set 3 is considerably smaller. This indicates that increase in thrust is not proportionate with mass flow rate increase. Higher characteristic velocity for set 2 and set 3 is indicative of larger increase of chamber pressure than increase of mass flow rate.

On the basis of the higher expansion ratio nozzle used we can draw the following conclusions. Observe Tables (28), (30) and (32).

The thrust F is increasing with increased mass flow rate. Specific impulse is improved for set 5 than for set 4 (except for reading 2), reason being thrust increases by larger amount than mass flow rate increase. But for set 6 once again the increase of mass flow rate dominates over the increase of thrust, resulting in lesser specific impulse than set 5. Thrust coefficient variation range is maximum for set 5 then comes set 6 and set 4. Characteristic velocity and equivalent velocity are higher for set 5 than for set 4 except for second reading in equivalent velocity comparision. Characteristic velocity and equivalent velocity are considerably smaller for set 6 than for set 5 (except for first reading), reason being the larger increase of mass flow rate than the increase of both chamber pressure and thrust.

For both the nozzles optimum performance (both specific impulse and characteristic velocity) is obtained by operating the rocket motor at lower O/F ratios, ofcourse at reduced nozzle performance i.e. thrust coefficient. For nozzle with $\epsilon = 1 : 1$ $A_t = \pi/4 \; D_t^2$ is greater than for nozzle with $\epsilon = 2.09 : 1$ $A_t$ by some 12%. The same can be the cause of reduced performance with 2nd nozzle. For this compare specific impulse and thrust coefficient for set 1 and 4 (exception first two readings in specific impulse comparison) and set 2 and 5.

# SECTION II

## THEORETICAL INVESTIGATION OF STATIC PERFORMANCE OF A CHEMICAL ROCKET MOTOR

# CHAPTER I

# INTRODUCTION

1.1 General

1.2 Work Related To Theoretical Performance Analysis

1.3 Present Work

## 1.1 General

Theoretical investigations of performance analysis of chemical rockets are of prime importance to guide propellant research by leading the lines of efforts towards high performance system, to aid engine design by providing information on temperature, pressure, composition of product gases and optimum mixture ratio and to provide data input for flight performance calculations. All these prehand data can be made available and a systematic theoretical approach to the problem of performance analysis is made. To handle the problem easily, appropriate assumptions are made.

## 1.2 Work Related to Theoretical Performance Analysis

In their report supported by NACA, Huff et al (19) attempted to devise a general method for computation of equilibrium composition and adiabatic temperature of chemical reaction with N number of components in product gas composition. Beighley (20) and Donegan (21) also tried to use computer systems. Donegan tried to determine thermochemical propellant calculations on high speed digital computers while Beighley concentrated on reduction of rocket motor performance data using IBM computing machines. All their individual efforts were towards more general system of propellants. Morgan (22) applied theoretical approach for thrust determination of rocket motor for C-H-N-O-F system. Gordon et al (23) worked on problem of theoretical performance analysis of chemical rocket motor using liquid oxygen and hydrogen.

In this technical note supported by NASA, Gordon (24) et al generalised methods for both equilibrium composition computation and for theoretical rocket performance. They developed computer programme in 'SOAP II', for the same. Moffatt et al (25) dealt with in particular $H_2$ - $O_2$ combination and determined product of combustion at elevated temperature. Mentz (26) developed the computer programme for the subject concerning equilibrium temperature and composition determination. He nicely discussed the convergence techniques for the problem. Stanely (27) developed an integrated programme for propellant performance analysis. All necessary data with graph required were arranged within his programme. Olson (28) studied both theoretically and experimentally the recombination and condensation process in high area ratio nozzles for various propellants. Many standard text books like Penner (29), Wilkins (30) have also dealt with this problem as a subject. Throughout Scarbrough's (31) book has been a useful reference for mathematics involved in the problem.

## 1.3 Present Work

The theoretical approach to the problem of performance analysis of a chemical rocket is initiated to have a method of computing from a given propellant combination of Hydrogen-Oxygen-Nitrogen system and chamber pressure; the adiabatic combustion temperature, equilibrium gas composition and other performance parameter from both frozen flow and equilibrium flow techniques. Ultimate aim is to know the optimum working conditions.

# CHAPTER II

## THEORETICAL CALCULATIONS

2.1 Determination of Combustion Temperature and Composition of Product Gas

2.2 Assumptions for Theoretical Performance Analysis of A Chemical Rocket and Introduction to Performance Parameters

2.3 Frozen Flow Approach

2.4 Equilibrium Flow Approach

## 2.1 Determination of Combustion Temperature and Composition of Product Gas

Consider the following basic equation for chemical reaction of $H_2 - O_2 - N_2$ system where $N_2$ is introduced as an impurity.

$$A\,(m\,H_2 + O_2 + K\,N_2) \rightleftharpoons n_1\,H_2O + n_2\,OH + n_3\,NO$$
$$+ \overset{n_4}{\cancel{N_4}}\,H_2 + n_5\,O_2 + n_6\,N_2 + n_7\,H + n_8\,O + n_9\,N \qquad (2.1.1)$$

where

$m = 16/OFR$ and

$K = ((16/OFR) + 1) \times 0.001$

$OFR = O/F$ ratio.

Aim is to determine $A$, $n_i$ $(i = 1, 2, \text{------} 9)$ and combustion temperature $T$. To simplify the calculations $A$ and $n_i$ are determined such that

$$p_i = n_i \qquad (2.1.2)$$

i.e. species volume is specified to be $\dot{R}\,T$. Applying Dalton's law of partial pressure we have for all gaseous phases

$$P = p_1 + p_2 + \text{----} + p_9 = n_1 + n_2 \text{----} + n_9 \qquad (2.1.3)$$

Law of conservation of mass is applied to the moles of species H

$$a = \frac{1}{A}\,(2\,n_1 + n_2 + 2\,n_4 + \cancel{2}\,n_7) = 2\,m \qquad (2.1.4)$$

for moles of species O

$$b = \frac{1}{A}\,(n_1 + n_2 + n_3 + 2\,n_5 + n_8) = 2 \qquad (2.1.5)$$

for modes of species N

$$c = \frac{1}{A} ( n_3 + 2 n_6 + n_9) = 2 K \qquad (2.1.6)$$

Also consider the following dissociation equations

$$H_2 + \frac{1}{2} O_2 \rightleftharpoons H_2 O \qquad R_1 = \frac{n_1}{n_4 (n_5)^{1/2}} \qquad (2.1.7)$$

$$H_2 O \rightleftharpoons \frac{1}{2} H_2 + O H \qquad R_2 = \frac{n_2 (n_4)^{1/2}}{n_1} \qquad (2.1.8)$$

$$\frac{1}{2} N_2 + \frac{1}{2} O_2 \rightleftharpoons N O \qquad R_3 = \frac{n_3}{(n_6)^{1/2} (n_5)^{1/2}} \qquad (2.1.9)$$

$$\frac{1}{2} H_2 \rightleftharpoons H \qquad R_4 = \frac{n_7}{(n_4)^{1/2}} \qquad (2.1.10)$$

$$\frac{1}{2} O_2 \rightleftharpoons O \qquad R_5 = \frac{n_8}{(n_5)^{1/2}} \qquad (2.1.11)$$

$$\frac{1}{2} N_2 \rightleftharpoons N \qquad R_6 = \frac{n_9}{(n_6)^{1/2}} \qquad (2.1.12)$$

Also for adiabatic combustion

$$m (H_o^T)_{H_2} + (H_o^T)_{O_2} + K (H_o^T)_{N_2} = \frac{1}{A} \left[ (H_o^T)_{H_2O} \; n_1 + (H_o^T)_{OH} \; n_2 + \right.$$
$$\left. ---- + (H_o^T)_N \; n_9 \right] \qquad (2.1.13)$$

Thus we have a system of eleven simultaneous nonlinear algebric equations from (2.1.3) to (2.1.13) for eleven unknowns to be determined, viz A, T and $n_i$ (i = 1, 2, ---- 9) for a given value of combustion pressure and O/F ratio.

Analytical solution of eleven nonlinear simultaneous algebric equations being extremely tedeous, suitable technique may be used to solve them numerically. Newton - Rapson's method of iteration is used to make the equations linear which are then solved using matrix inversion method by Gauss - Jordan and next iteration is started if any. To speed up the iteration process natural logarithm system is used.

Consider equation (2.1.4). Take the natural logarithm of both the sides

$$\therefore \quad \ln a = \ln (2 n_1 + n_2 + 2 n_4 + n_7) - \ln A \qquad (2.1.14)$$

Considering first order variations we have

$$\Delta \ln a = \frac{\partial \ln a}{\partial \ln n_1} \Delta \ln n_1 + \frac{\partial \ln a}{\partial \ln n_2} \Delta \ln n_2 + \frac{\partial \ln a}{\partial \ln n_4} \Delta \ln n_4 + \frac{\partial \ln a}{\partial \ln n_7} \Delta \ln n_7 + - \frac{\partial \ln a}{\partial \ln A} \Delta \ln A \qquad (2.1.15)$$

also $$\frac{\partial \ln a}{\partial \ln n_1} = \frac{\partial \ln a}{\partial n_1} \cdot \frac{\partial n_1}{\partial \ln n_1} = \frac{\partial \ln a}{\partial n_1} \cdot \frac{1}{\frac{\partial \ln n_1}{\partial n_1}}$$

$$= \left( \frac{2}{2 n_1 + n_2 + 2 n_4 + n_7} \right) n_1 = \frac{2 n_1}{(a\ A)_{Calc}}$$

Proceeding similarly we have

$$\frac{\partial \ln a}{\partial \ln n_2} = \frac{n_2}{(a\ A)_{Calc}} \quad ; \quad \frac{\partial \ln a}{\partial \ln n_4} = \frac{2 n_4}{(a\ A)_{Calc}}$$

$$\frac{\partial \ln a}{\partial \ln n_7} = \frac{n_7}{(a\ A)_{Calc}} \quad ; \quad \frac{\partial \ln a}{\partial \ln A} = -1 \text{ etc}$$

also $\Delta \ln a = \ln a_{True} - \ln a_{Calc} = \ln \left( \frac{a_{True}}{a_{Calc}} \right)$

Thus Eq. (2.1.15) becomes

$$\therefore (a\ A)_{Calc} \ln \left( \frac{a_{Ture}}{a_{Calc}} \right) = 2\, n_1\ \Delta \ln n_1 + n_2\ \Delta \ln n_2 + 2\, n_4\ \Delta \ln n_4 + n_7\ \Delta \ln n_7 - (a\ A)_{Calc}\ \Delta \ln A \tag{2.1.16}$$

Similarly rest of the equations can be linearised.

They become

$$(b\ A)_{Calc} \ln \left( \frac{b_{Ture}}{b_{Calc}} \right) = n_1\ \Delta \ln n_1 + n_2\ \Delta \ln n_2 + n_3\ \Delta \ln n_3 + 2\, n_5\ \Delta \ln n_5 + n_8\ \Delta \ln n_8 - (b\ A)_{Calc}\ \Delta \ln A \tag{2.1.17}$$

$$(c\ A)_{Calc} \ln \left( \frac{c_{True}}{c_{Calc}} \right) = n_3\ \Delta \ln n_3 + 2\, n_6\ \Delta \ln n_6 + n_9\ \Delta \ln n_9 - (c\ A)_{Calc}\ \Delta \ln A \tag{2.1.18}$$

$$P_{Calc} \ln \left( \frac{P_{True}}{P_{Calc}} \right) = \sum_{i=1}^{i=9} n_i\ \Delta \ln n_i \tag{2.1.19}$$

$$\ln \left( \frac{R_{1\ True}}{R_{1\ Calc}} \right) = \Delta \ln n_1 - \Delta \ln n_4 - \frac{1}{2}\ \Delta \ln n_5 \tag{2.1.20}$$

$$\ln \left( \frac{R_{2\ True}}{R_{2\ Calc}} \right) = \Delta \ln n_2 + \frac{1}{2}\ \Delta \ln n_4 - 1 \cdot \Delta \ln n_1 \tag{2.1.21}$$

$$\ln \left( \frac{R_{3\ True}}{R_{3\ Calc}} \right) = \Delta \ln n_3 - \frac{1}{2}\ \Delta \ln n_5 - \frac{1}{2}\ \Delta \ln n_6 \tag{2.1.22}$$

$$\ln \left( \frac{R_{4\ True}}{R_{4\ Calc}} \right) = -\frac{1}{2} \Delta \ln n_4 + \Delta \ln n_7 \qquad (2.1.23)$$

$$\ln \left( \frac{R_{5\ True}}{R_{5\ Calc}} \right) = -\frac{1}{2} \Delta \ln n_5 + \Delta \ln n_8 \qquad (2.1.24)$$

$$\ln \left( \frac{R_{6\ True}}{R_{6\ Calc}} \right) = -\frac{1}{2} \Delta \ln n_6 + \Delta \ln n_9 \qquad (2.1.25)$$

$$(h\ A)_{Calc} \ln \left( \frac{h_{True}}{h_{Calc}} \right) = \sum_{i=1}^{i=9} (H_o^T)_i\ n_i\ \Delta \ln n_i -$$

$$(h\ A)_{Calc}\ \Delta \ln A + \sum_{i=1}^{i=9} c^o_{p_i}\ n_i\ T\ \Delta \ln T \qquad (2.1.26)$$

where

$$(H_o^T)_i = \left[ \int_o^T C_p^o\ dT + H_o^o \right]_i$$

Note : P = $p_c$ and T = $T_c$ here.

Thus modified equations from (2.1.16) to (2.1.26) are to be solved for $\Delta \ln n_i$'s, $\Delta \ln A$ and $\Delta \ln T$. Matrix form of the above equations is represented as on the next page.

$$
\begin{bmatrix}
2n_1 & n_2 & 0 & 2n_4 & 0 & 0 & n_7 & 0 & 0 & (-aA)_{calc} & 0 \\
n_1 & n_2 & n_3 & 0 & 2n_5 & 0 & 0 & n_8 & 0 & (-bA)_{calc} & 0 \\
0 & 0 & n_3 & 0 & 0 & 2n_6 & 0 & 0 & n_9 & (-cA)_{calc} & 0 \\
n_1 & n_2 & n_3 & n_4 & n_5 & n_6 & n_7 & n_8 & n_9 & 0 & 0 \\
1 & 0 & 0 & -1 & -1/2 & 0 & 0 & 0 & 0 & 0 & 0 \\
-1 & 1 & 0 & 1/2 & 0 & 0 & 0 & 0 & 0 & 0 & 0 \\
0 & 0 & 1 & 0 & -1/2 & -1/2 & 0 & 0 & 0 & 0 & 0 \\
0 & 0 & 0 & -1/2 & 0 & 0 & 1 & 0 & 0 & 0 & 0 \\
0 & 0 & 0 & 0 & -1/2 & 0 & 0 & 1 & 0 & 0 & 0 \\
0 & 0 & 0 & 0 & 0 & -1/2 & 0 & 0 & 1 & 0 & 0 \\
H^T_{01}n_1 & H^T_{02}n_2 & H^T_{03}n_3 & H^T_{04}n_4 & H^T_{05}n_5 & H^T_{06}n_6 & H^T_{07}n_7 & H^T_{08}n_8 & H^T_{09}n_9 & (-hA)_{calc} & (\sum_i C^\circ_{p_i} n_i)T
\end{bmatrix}
\begin{bmatrix}
\Delta \ln n_1 \\ \Delta \ln n_2 \\ \Delta \ln n_3 \\ \Delta \ln n_4 \\ \Delta \ln n_5 \\ \Delta \ln n_6 \\ \Delta \ln n_7 \\ \Delta \ln n_8 \\ \Delta \ln n_9 \\ \Delta \ln A \\ \Delta \ln T
\end{bmatrix}
=
\begin{bmatrix}
(aA)_{calc} \ln\left(\frac{a_{True}}{a_{calc}}\right) \\
(bA)_{calc} \ln\left(\frac{b_{True}}{b_{calc}}\right) \\
(cA)_{calc} \ln\left(\frac{c_{True}}{c_{calc}}\right) \\
P_{calc} \ln\left(\frac{P_{True}}{P_{calc}}\right) \\
\ln\left(\frac{R_1\ True}{R_1\ Calc}\right) \\
\ln\left(\frac{R_2\ True}{R_2\ Calc}\right) \\
\ln\left(\frac{R_3\ True}{R_3\ calc}\right) \\
\ln\left(\frac{R_4\ True}{R_4\ Calc}\right) \\
\ln\left(\frac{R_5\ True}{R_5\ Calc}\right) \\
\ln\left(\frac{R_6\ True}{R_6\ calc}\right) \\
(hA)_{calc} \ln\left(\frac{h_{True}}{h_{Calc}}\right)
\end{bmatrix}
\tag{2.1.27}
$$

For numerical solution it is essential that diagonal elements should be non zero. Necessary care is taken in arranging the system as shown in Appendix B.1.

To start with the solution of the above system assume initial guess for $n_i$, A and T. Calculate all true thermodynamic quantities at temperature T. Calculated values are found from the formulation. After finding the solutions, which are being error terms, correct the assumed values of $n_i$, A and T, till the maximum error is some minimum quantity. Note that $n_j = n_i/A$ since value of A was so calculated to simplify the solution. Data, sample calculations, computer programme with necessary modifications for numerical solution and calculated data are represented in Appneix B.1.

## 2.2 Assumptions for Theoretical Performance Analysis of A Rocket Motor and Introduction to Performance Paramenters (See Penner (29))

Analysis is based on the following assumptions.

(1) Combustion process is adiabatic and after it, thermodynamic equilibrium is assumed to be reached.

(2) Products of combustion behave as ideal gases.

(3) Expansion of products of combustion in a nozzle can be considered as one dimensional flow of non viscous, ideal gases.

(4) Expansion again is assumed to be adiabatic.

(5) Velocity at the entrance of expansion nozzle is negligibly small in comparision to exit velocity from the nozzle.

(6) Particular average value of $\bar{\gamma}$ and $\bar{W}$ is assumed.

With the above assumptions, following parameters for a chemical rocket motor can be defined. We will restrict to a

case of atmospheric expansion in a small divergence angle nozzle.

Thrust $F = \frac{1}{2}(1 + \cos\alpha)\,\dot{w}\,\frac{v_e}{g} + (p_e - p_a)\,A_e$ (2.2.1)

$$F = \dot{w}\,u_{eq}/g \qquad (2.2.2)$$

where $u_{eq}$ is Equivalent or Effective exhaust velocity.

$$u_{eq} = \frac{1}{2}(1 + \cos\alpha)\,v_e + \frac{(p_e - p_a)\,A_e}{\dot{w}/g} \qquad (2.2.3)$$

for $p_e = p_a$ and $\alpha$ very small we have

$$u_{eq} = v_e \qquad (2.2.4)$$

$$F = \dot{w}\,v_e/g \qquad (2.2.5)$$

Specific Impulse $I_{sp} = F/\dot{w} = u_{eq}/g$ (2.2.6)

for $p_e = p_a$ and $\alpha$ very small we have

$$I_{sp} = v_e/g \qquad (2.2.7)$$

Also $T_e/T_c = (p_e/p_c)^{\frac{\bar{\gamma}^* - 1}{\bar{\gamma}^*}}$ (2.2.8)

Characteristic velocity $C^* = \frac{1}{\Gamma^*}\left(\frac{g\,J\,\dot{R}\,T_c}{\bar{W}^*\;\bar{\gamma}^*}\right)$ (2.2.9)

where (2.2.10)

$$\Gamma^* = \left(\frac{2}{\bar{\gamma}^* + 1}\right)^{\frac{\bar{\gamma}^* + 1}{2(\bar{\gamma}^* - 1)}}$$

Thrust Coefficient $C_F = u_{eq} / C^*$ (2.2.11)

$$F = C_F\,p_c\,A_t \qquad (2.2.12)$$

Particulat average value of $\bar{\gamma}$ and $\bar{W}$ is taken as

$$\gamma_c = \gamma_e = \bar{\gamma} = \bar{\gamma}^* \quad \text{and} \tag{2.2.13}$$

$$W_c = W_e = \bar{W} = \bar{W}^* \tag{2.2.14}$$

## 2.3 Frozen Flow Approach

Here recombination process occurs in a time which is relatively larger in comparision to the expansion time. That is to say that reaction rates are very slow and the propellant products are said to be frozen at combustion chamber composition.

Applying law of conservation of energy at entry and exit of rocket nozzle

$$\frac{1}{2}\,\bar{W}\,v_e^2 = \Delta H_c^e \tag{2.3.1}$$

where

$$\bar{W} = \sum_{j=1}^{j=9} X_j\,W_j \quad \text{and} \quad X_j = n_j / n_{Total} \tag{2.3.2}$$

also

$$\Delta H_c^e = \sum_{j=1}^{j=9} X_j\,(H_{j,T_c} - H_{j,T_e}) \tag{2.3.3}$$

Now to determine $T_e$, we will assume the flow through nozzle as isentropic flow though adiabatic flow will give the same results as developed by Penner (29). Thus for isentropic flow through nozzle we have

$$\bar{S}_{T_c,\,p_c} = \bar{S}_{T_e,\,p_e} \tag{2.3.4}$$

for mixture of ideal gases Penner (29) gives

$$\bar{S}_{T,p} = \sum_{j=1}^{j=9} X_j \, S^o_{j,T} - \dot{R} \left[ \ln p + \sum_{j=1}^{j=9} X_j \ln X_j \right] \quad (2.3.5)$$

From (2.3.4) and (2.3.5) we have

$$\sum_{j=1}^{j=9} X_j \left( S^o_{j,T_c} - S^o_{j,T_e} \right) = \dot{R} \ln p_c/p_e \quad (2.3.6)$$

Brief procedure for performance parameter determination is outlined here.

(1) Find combustion temperature $T = T_c$ and composition using the theory developed in 2.1.

(2) For $p_e = p_a = 1$ ata, find $T_e$ by trial and error method using Eq. (2.3.6).

(3) Find $\bar{W}$ using Eq. (2.3.2)

(4) Find enthalpy change during expansion using Eq. (2.3.3)

(5) Find $v_e$ using Eq. (2.3.1) for small $\alpha$ and $p_e = p_a$

(6) Find $I_{sp}$ using Eq. (2.2.7) for small $\alpha$ and $p_e = p_a$

(7) Find $\bar{\gamma}^*$ using Eq. (2.2.8)

(8) Find C* using Eq. (2.2.9) and (2.2.10)

(9) Find $C_F$ using Eq. (2.2.11)

All these calculations are for a fixed combustion chamber pressure $p_c$ and for fixed O/F ratio. The same is repeated for various $p_c$'s and O/F ratios. Data, sample calculations, computer programme and calculated readings are given in Appendix B.2.

## 2.4 Equilibrium Flow Approach

Here reaction rates are too fast and various recombination processes occur in a time which is short in comparision to expansion time. Thus corresponding to each temperature thermo-chemical equilibrium is achieved and new composition determination is a must (or reaction rates for both forward and backward reactions are same at each temperature encountered in expansion flow).

Equation (2.3.1) is still valid for equilibrium flow analysis except that one must restrict the analysis to a given weight of gas mixture, rather than to the temperature dependent molecular weight of gases. Choosing $\overline{W}_c$ as initial molecular weight and then applying law of conservation of energy at nozzle entry and exit we have

$$\frac{1}{2}\ \overline{W}_c\ v_e^2 = \overline{H}_c - \overline{H}_e \left( \frac{\overline{W}_c}{\overline{W}_e} \right) \tag{2.4.1}$$

where

$$\overline{W}_c = \sum_{j=1}^{j=9} X_{j,\ T_c,\ p_c}\ W_j \qquad \& \tag{2.4.2}$$

$$\overline{W}_e = \sum_{j=1}^{j=9} X_{j,\ T_e,\ p_e}\ W_j \tag{2.4.3}$$

For $\overline{H}_c$ and $\overline{H}_e$, absolute molar enthalpies of the gas mixture referred to 298.16° K as reference point. Thus

$$\overline{H}_c = \sum_{j=1}^{j=9} X_{j,\ T_c\ p_c} \left[ \Delta Hf_j^o + H_{j,\ T_c} - H_{j,298.16} \right] \tag{2.4.4}$$

and

$$\overline{H}_e = \sum_{j=1}^{j=9} X_{j,\,T_e,\,p_e} \left[\Delta Hf^o_j + H_{j,\,T_e} - H_{j,298.16}\right] \tag{2.4.5}$$

The ratio $\overline{W}_c / \overline{W}_e$ in (2.4.1) corrects the molar enthalpy at nozzle exit to the enthalpy for the same weight of gas mixture for which $\overline{H}_c$ is calculated.

For equilibrium flow the only method utilising the condition that the expansion is isentropic, is used for determination of $T_e$. Isentropic flow occurs for total weight of the gas and not for temperature dependent molecular weight of the mixture. Thus

$$\overline{S}_{T_c,\, p_c} = (\overline{W}_c/\overline{W}_e)\, \overline{S}_{T_e,p_e} \tag{2.4.6}$$

using Eq. (2.4.6) and Eq. (2.3.5) we have

$$\sum_{j=1}^{j=9} X_{j,\,T_c,\,p_c} \left( S^o_{j,T_c} - \dot{R} \ln p_c - \dot{R} \ln X_{j,T_c,p_c}\right)$$

$$= (\overline{W}_c/\overline{W}_e) \sum_{j=1}^{j=9} X_{j,T_e,p_e} \left( S^o_{j,T_e} - \dot{R} \ln p_e - \dot{R} \ln X_{j,T_e,p_e}\right) \tag{2.4.7}$$

which can be further simplified to

$$\sum_{j=1}^{j=9} \left\{ X_{j,T_c,p_c} (S^o_{j,T_c} - \dot{R} \ln X_{j,T_c,p_c}) - (\overline{W}_c/\overline{W}_e) x \; X_{j,T_e,p_e} (S^o_{j,T_e} - \dot{R} \ln X_{j,T_e,p_e}) \right\}$$

$$= \dot{R} \left[\ln p_c - (\overline{W}_c/\overline{W}_e) \ln p_e\right] \tag{2.4.8}$$

Equation (2.4.8) can be used to find $T_e$ by trial and error method. For every $T_e$, $p_e$ set; $X_{j,T_e,p_e}$ is to be determined. Also $\bar{\gamma}^*$ has no meaning in shifting equilibrium flow. Though it can be approximated by Eq. (2.2.8) also $\bar{W}^*$ is modified to

$$\bar{W}^* = \frac{1}{2} ( \bar{W}_c + \bar{W}_e) \qquad (2.4.9)$$

Brief procedure is outlined here.

(1) Find $T_c$ and composition using the theory developed in 2.1

(2) For $p_e = p_a$, calculate $T_e$ by trial and error method. For each assumed value of $T_e$ find composition i.e. $X_{j,T_e,p_e}$.

(3) Find enthalpy change using Eqs. (2.4.1), (2.4.4) and (2.4.5) by

$$\bar{H}_c - \left(\frac{\bar{W}_c}{\bar{W}_e}\right) \bar{H}_e = \sum_{j=1}^{j=9} \Bigg\{ X_{j,T_c,p_c} \left( \Delta Hf_j^o + H_{j,T_c} - H_{j,298.16}\right) - \left( \frac{\bar{W}_c}{\bar{W}_e} \right) X_{j,T_e,p_e} \left( \Delta Hf_j^o + H_{j,T_e} - H_{j,298.16}\right)\Bigg\} \qquad (2.4.10)$$

(4) Find $v_e$ using Eq. (2.4.1) and (2.4.10)

(5) For $p_e = p_a$ and small $\alpha$ find $I_{sp}$ using Eq. (2.2.7)

(6) Find $\bar{\gamma}^*$ using Eq. (2.2.8)

(7) Find C* using Eqs. (2.2.9), (2.2.10) and (2.4.9)

(8) Find $C_F$ using Eq. (2.2.11)

All these calculations are for a fixed combustion chamber pressure and O/F ratio. The same is repeated for various $p_c$'s and O/F ratios. Data, sample calculations, computer programme and calculated data is given in Appendix B.3.

# CHAPTER III

## RESULTS AND DISCUSSION

3.1 Analysis of Combustion Temperature and Composition of Product Gas

3.2 Frozen Flow Approach

3.3 Equilibrium Flow Approach

## 3.1 Analysis of Combustion Temperature and Composition of Product Gas

The plots of the data given in the Tables (33), (34), (35) and (36); (37), (38), (39) and (40); (41), (42), (43) and (44); (45), (46), (47) and (48) are given in Figures (27), (28), (29) and (30). These figures show the variations of combustion temperature and mole fractions of various species assumed to be present in the product gas, for combustion chamber pressures varying from 50 Psia to 200 Psia, in step of 50 Psia and O/F ratio 2 to 8, in step of 2 each.

From the figures (27), (28), (29) and (30), for a particular set following can be concluded when O/F ratio is increasing.

(1) Equilibrium combustion temperature increases. At higher O/F ratio this increase is flatten.

(2) Mole fraction for water increases. At higher O/F ratio this increment is reduced.

(3) Mole fraction for OH increases.

(4) Mole fraction for NO increases.

(5) Mole fraction for $H_2$ decreases.

(6) Mole fraction for $O_2$ increases.

(7) Mole fraction for $N_2$ decreases.

(8) Mole fraction for H first increases and then decreases.

(9) Mole fraction for O increases.

(10) Mole fraction for N first increases and then decreases.

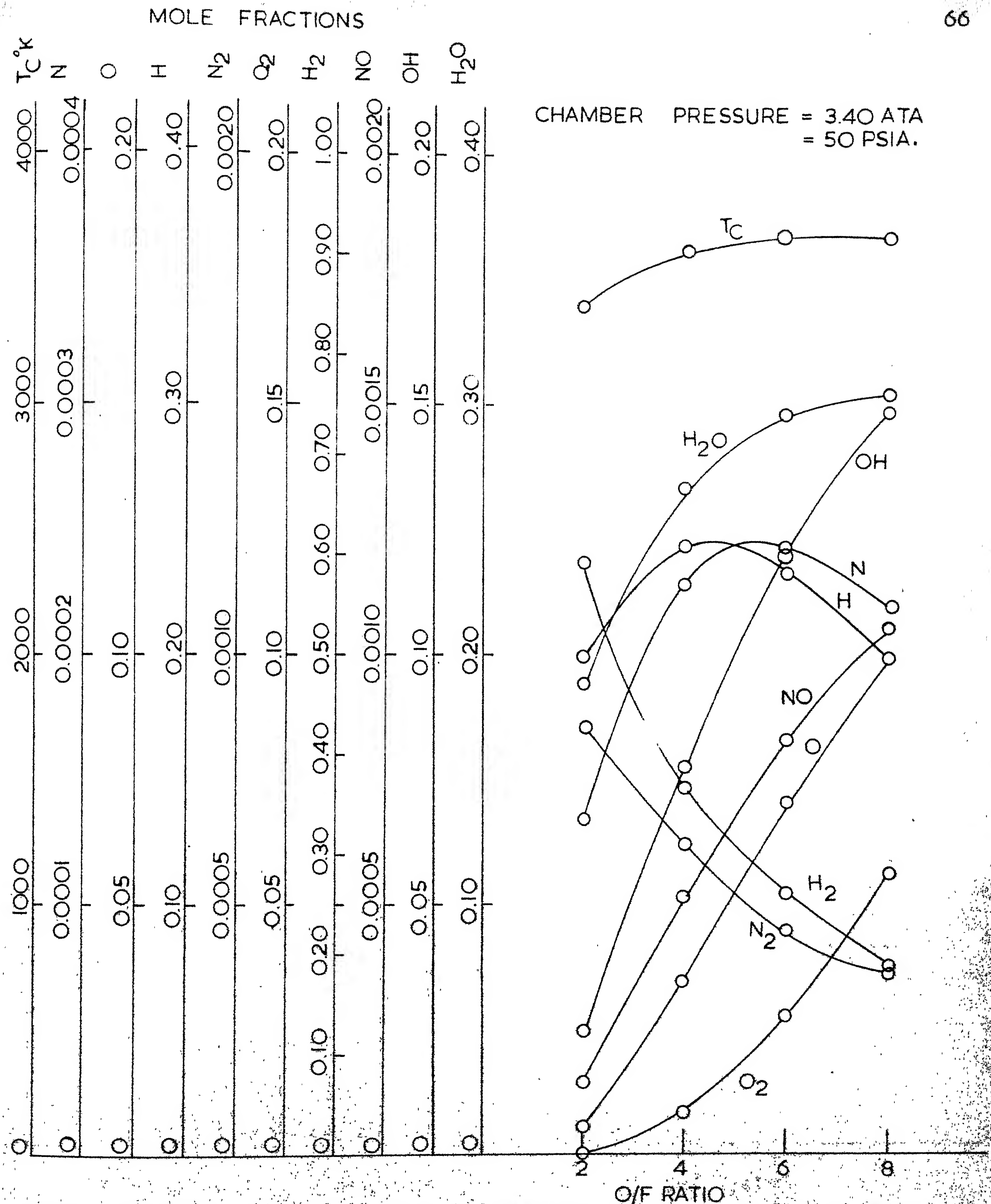


FIG.27_VARIATION OF THEORETICALLY CALCULATED EQUILIBRIUM TEMPERATURE AND PRODUCT GAS COMPOSITION FOR HYDROGEN _OXYGEN _NITROGEN PROPELLANT SYSTEM CHEMICAL ROCKET , WITH O/F RATIO

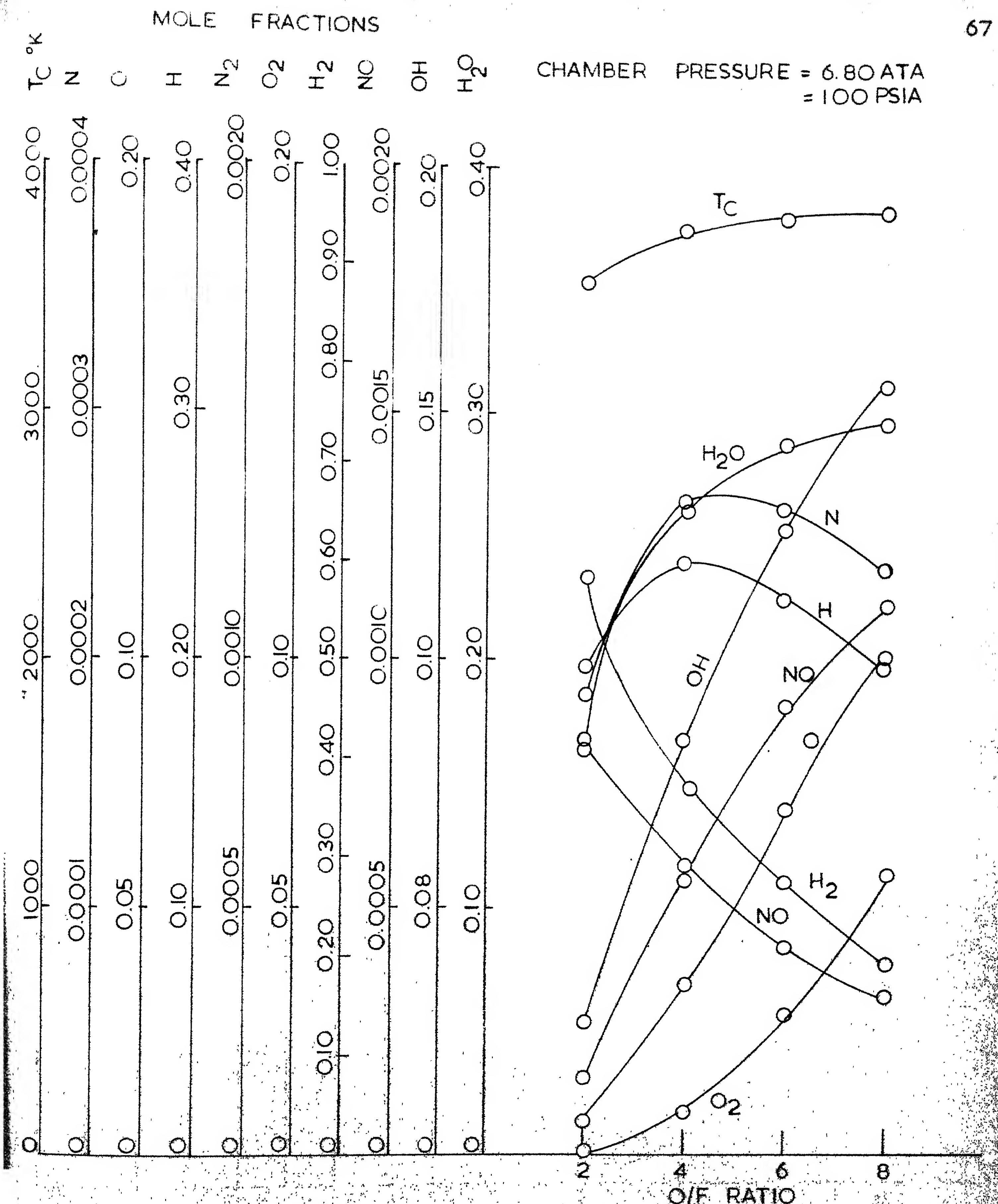


FIG. 28 _ VARIATION OF THEORETICALLY CALCULATED EQUILIBRIUM TEMPERATURE AND PRODUCT GAS COMPOSITION FOR HYDROGEN - OXYGEN - NITROGEN PROPELLANT SYSTEM CHEMICAL ROCKET, WITH O/F RATIO

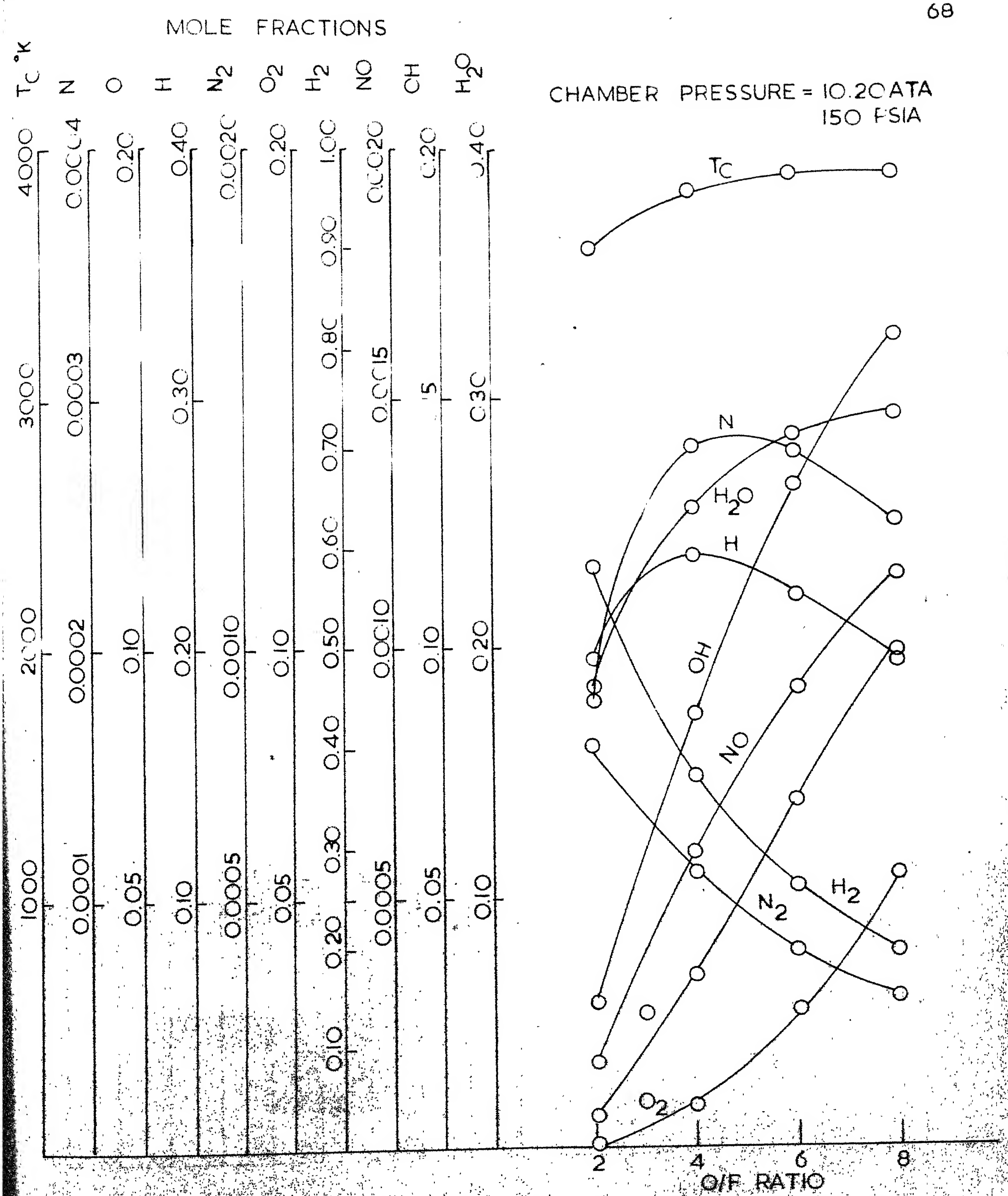


FIG.29. VARIATION OF THEORETICALLY CALCULATED EQUILIBRIUM TEMPERATURE AND PRODUCT GAS COMPOSITION FOR HYDROGEN - OXYGEN - NITROGEN PROPELLANT SYSTEM CHEMICAL ROCKET, WITH O/F RATIO

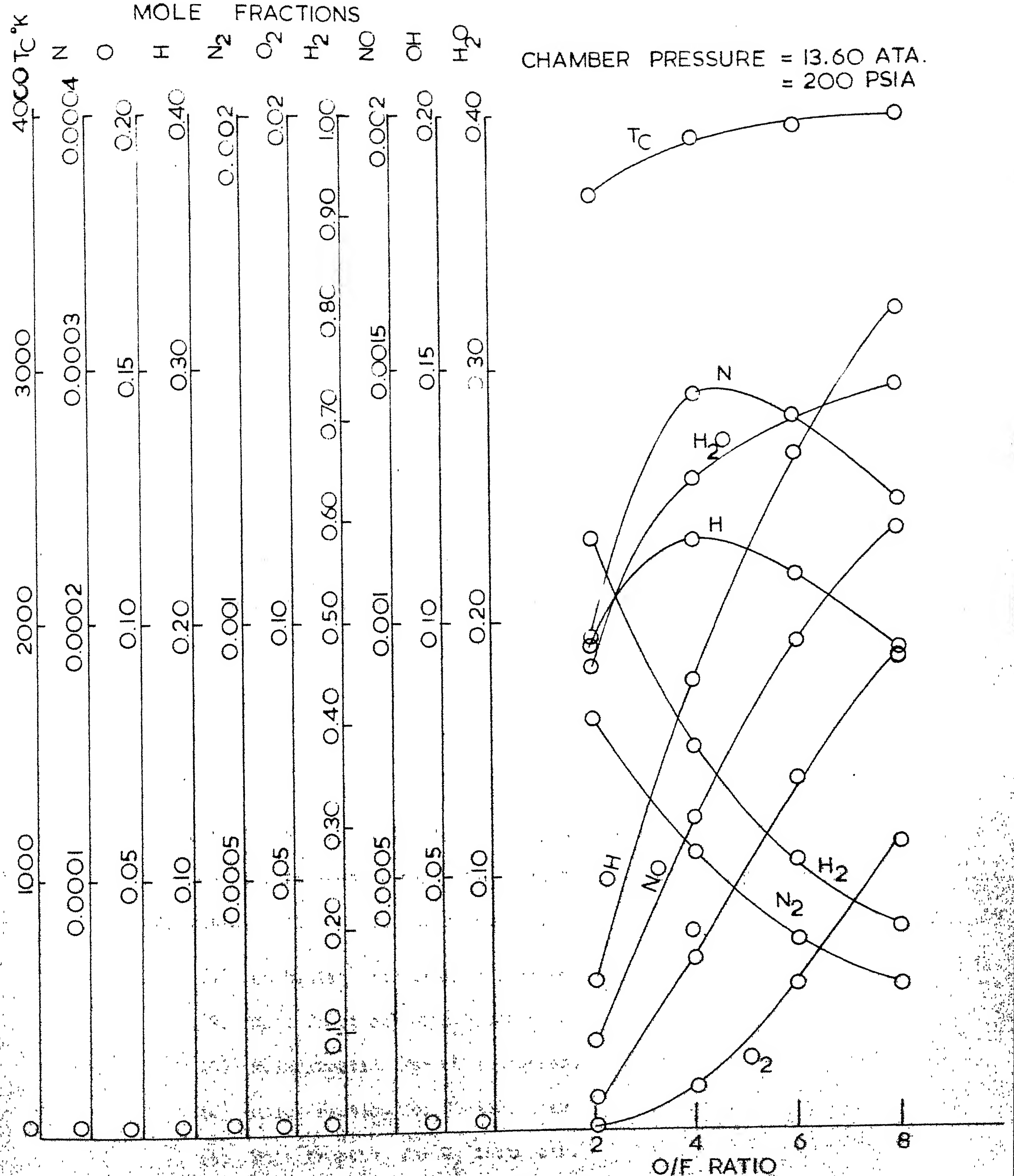


FIG. 30 – VARIATION OF THEORETICALLY CALCULATED EQUILIBRIUM TEMPERAT AND PRODUCT GAS COMPOSITION FOR HYDROGEN – OXYGEN – NITROG PROPELLANT SYSTEM CHEMICAL ROCKET, WITH O/F RATIO

Conclusions which include the increase of quantity can be explained as under. Availability of more oxygen with increased O/F ratio tends to complete combustion resulting in increased quantities of $T_c$ and mole fractions of $H_2O$, OH, NO, $O_2$, O etc. Mole fraction of $H_2$ decreases with O/F ratio increase because lesser amount of $H_2$ is available for taking part in reaction. Mole fraction of $N_2$ decreases as it is accounted for in factor K, in formulation which decreases with O/F ratio. Thus lesser available $N_2$, results in lesser $N_2$ in product gas. Dissociated mole fractions of N and H increases with O/F ratio at lower values of O/F ratios but at higher values of O/F ratio they decrease with increase in O/F ratios. Indicative of relative supression of dissociation of $N_2$ and $H_2$ at higher O/F ratio. Higher value of combustion temperature is due to the inclusion of all basic nine species, in particular NO, O and N, in formulation of the problem.

From the Table group (33), (37), (41) and (45); (34), (38), (42) and (46); (35), (39), (43), and (47); (36), (40), (44) and (48) for increasing combustion chamber pressure, at particular O/F ratio, following can be concluded.

(1) Combustion temperature increases.

(2) Mole fraction for $H_2O$ decreases.

(3) Mole fraction for OH increases.

(4) Mole fraction for NO increases.

(5) Mole fraction for $H_2$ increases.

(6) Mole fraction for $N_2$ decreases.

(7) Mole fraction for H decreases.

(8) Mole fraction for N increases.

(9) Variations in mole fractions of $O_2$ and O are bit non predictable.

Sole reason being the increase in combustion temperature with O/F ratio.

3.2 Frozen Flow Approach

The plots of data given in the Tables (49), (50), (51) and (52) are given in the figure (31), (32), (33) and (34). These figures include the variations of performance parameters of chemical rocket with $H_2$-$O_2$-$N_2$ propellant system for combustion chamber pressures 50, 100, 150 and 200 Psia and O/F ratios 2, 4, 6 and 8 at each chamber pressure.

From the figures (31), (32), (33) and (34), for a particular set, following can be concluded. When O/F ratio is increasing

(1) Exit temperature increases.

(2) Average molecular weight increases.

(3) Exit velocity decreases.

(4) Equivalent velocity decreases.

(5) Specific impulse decreases.

(6) Trend for $\overline{\gamma}^*$ and thrust coefficient being little bit non predictable.

(7) Characteristic velocity decreases.

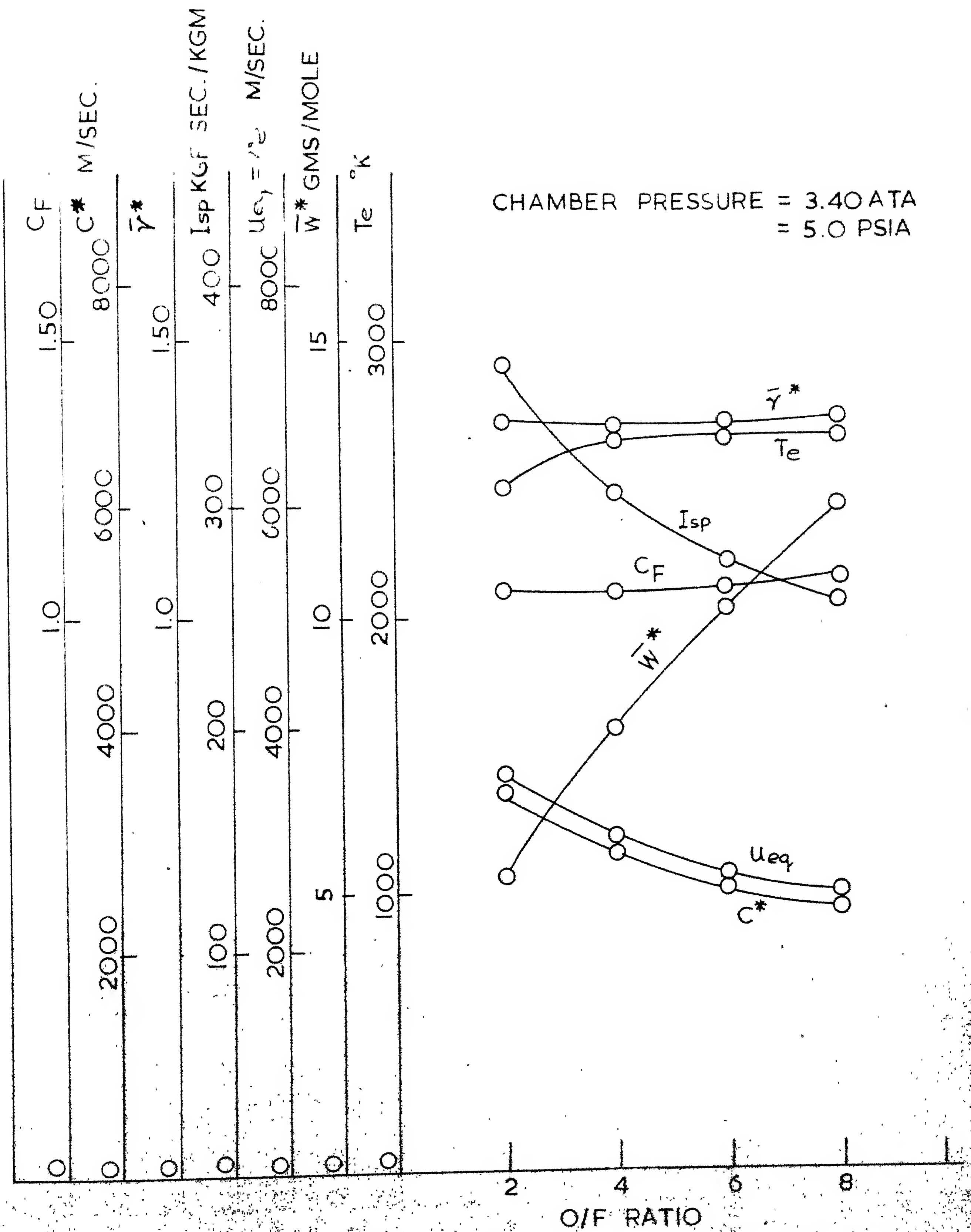


FIG. 31 – VARIATION OF THEORETICALLY CALCULATED PERFORMANCE PARAMETERS FOR HYDROGEN–OXYGEN–NITROGEN PROPELLANT SYSTEM CHEMICAL ROCKET; WITH O/F RATIO; WITH FROZEN FLOW APPROACH

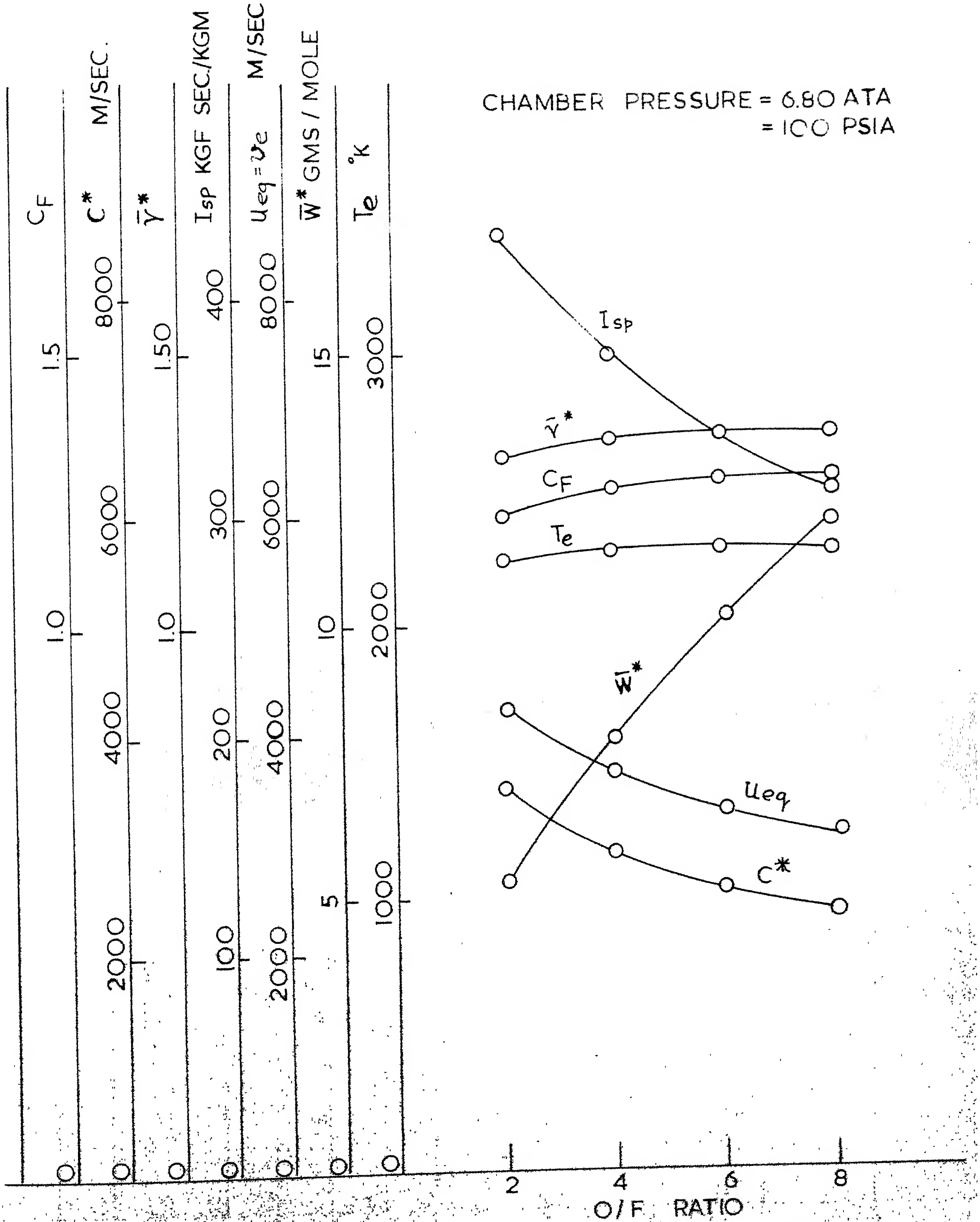


FIG.32 VARIATION OF THEORETICALLY CALCULATED PERFORMANCE PARAMETERS FOR HYDROGEN - OXYGEN - NITROGEN PROPELLANT SYSTEM CHEMICAL ROCKET, WITH O/F RATIOS, WITH FROZEN FLOW APPROACH

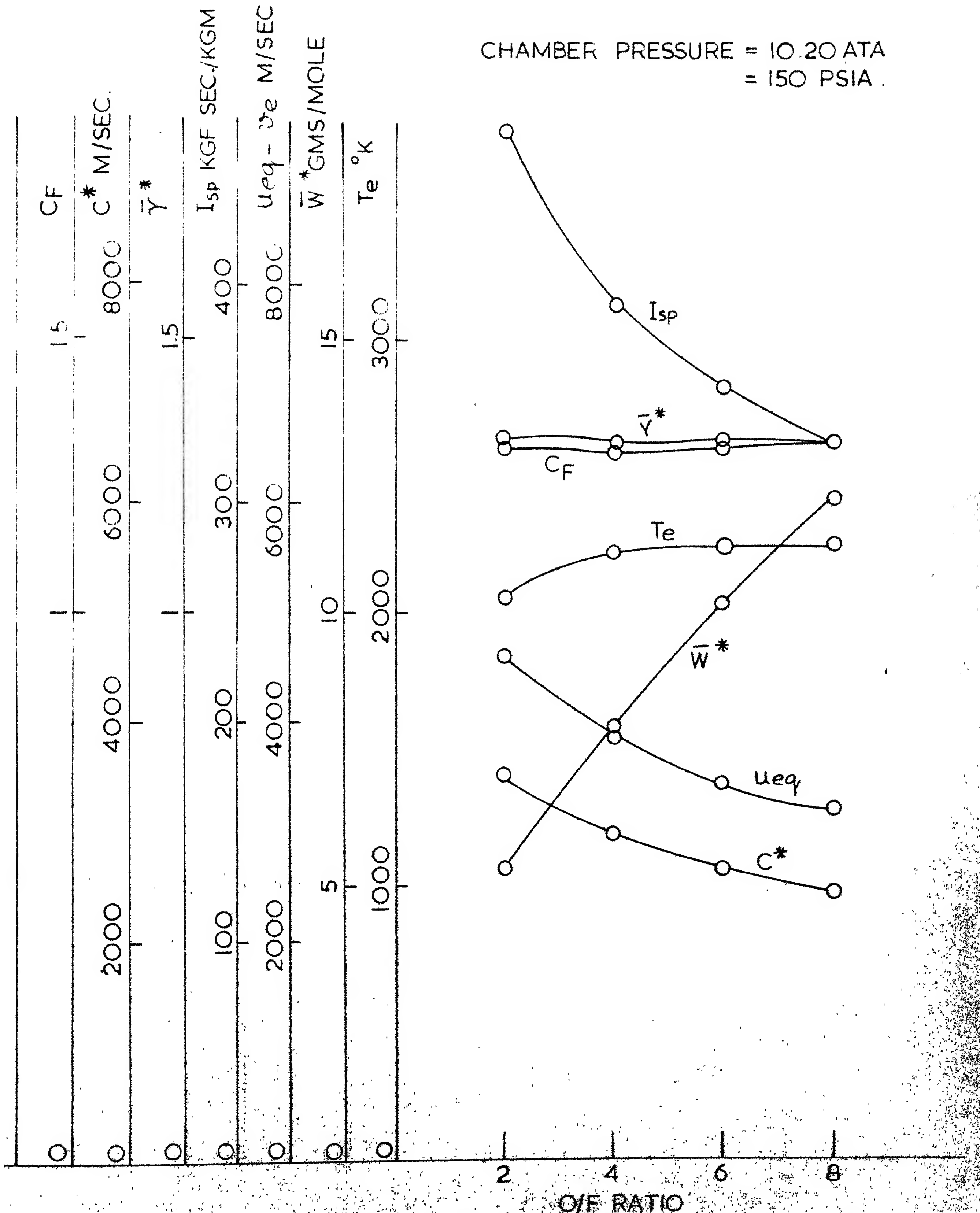


FIG. 33_ VARIATION OF THEORETICALLY CALCULATED PERFORMANCE PARAMETERS FOR HYDROGEN-OXYGEN-NITROGEN PROPELLANT SYSTEM CHEMICAL ROCKET WITH O/F RATIOS, WITH FROZEN FLOW APPROACH

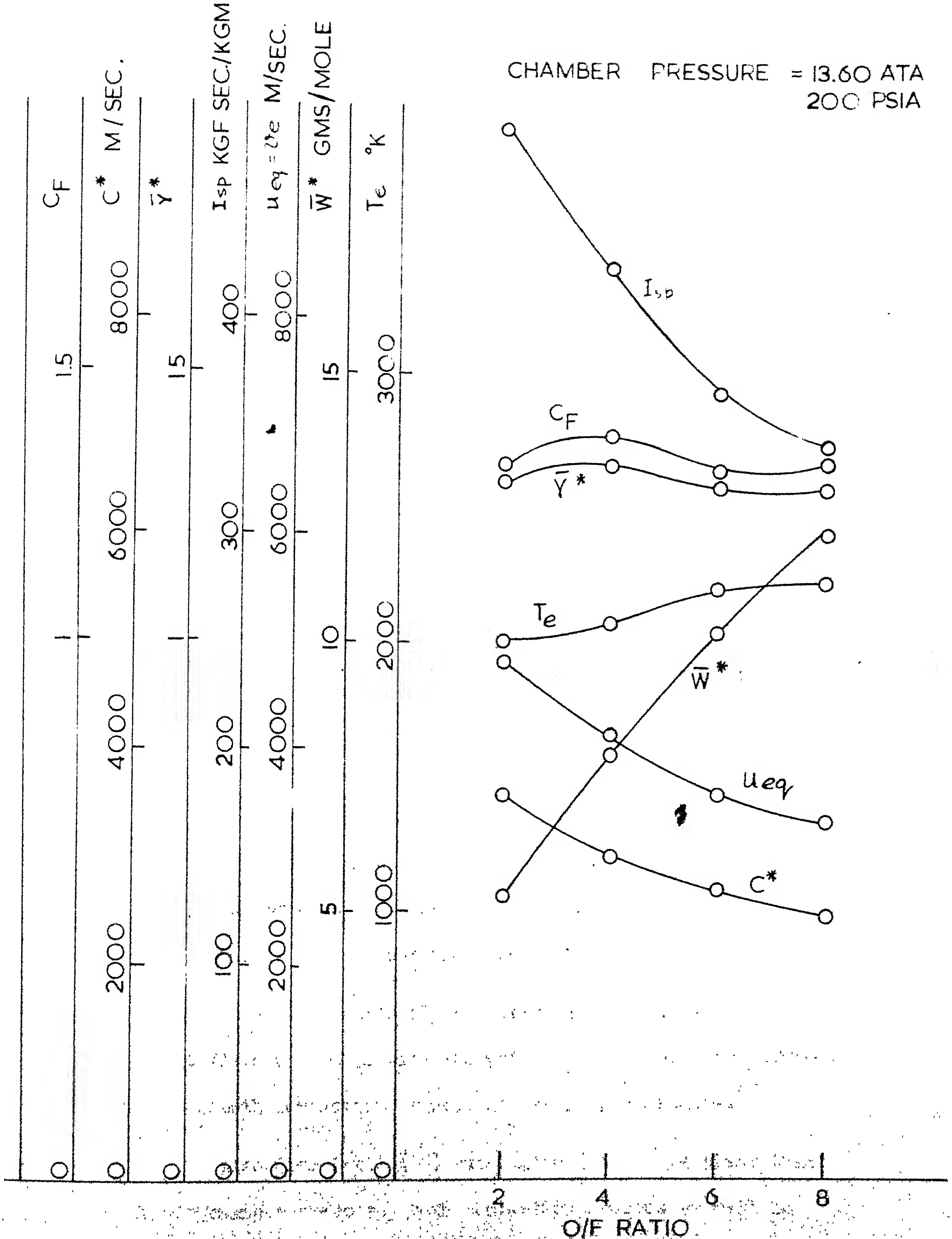


FIG. 34_ VARIATION OF THEORETICALLY CALCULATED PERFORMANCE PARAMETERS FOR HYDROGEN_OXYGEN_NITROGEN PROPELLANT SYSTEM CHEMICAL ROCKET, WITH O/F RATIOS ; WITH FROZEN FLOW APPROACH

The conclusions can be explained as under. With increased O/F ratio combustion temperature increases and nozzle being unchanged, exit temperature increases. Average molecular weight increases with increased and adjusted mole fractions of various species. Combined effect of above two reasons may be argued as the cause of conclusions (3), (4), (5) and (7).

From the tables (49), (50), (51) and (52) following can also be concluded. For increasing combustion chamber pressure and at fixed O/F ratio.

(1) Exit temperature decreases.

(2) Average molecular weight does not get affected a lot or it is approximately constant.

(3) Both linear exit velocity and equivalent velocity increases.

(4) Specific impulse increases.

(5) $\bar{\gamma}^*$ variations are bit less predictable but it generally decreases.

(6) Characteristic velocity increases.

(7) Thrust coefficient increases.

Thus it is obviously advantageous to run the rocket motor at fixed O/F ratio with increased $p_c$ to obtain more efficiency of both combustion as well as nozzle performance.

Also from Table (49) and Figure (31) it is clear that at lower chamber pressure, both combustion process as well as nozzle performance are not efficient. This is obvious and is

indicated by lower values of both thrust coefficient and difference between equivalent velocity and characteristic velocity. Higher values of exit temperature and lower values of specific impulse for this case are also indicative of the same.

## 3.3 Equilibrium Flow Approach

The plots of data given in the Tables (53), (54), (55) and (56) are given in the Figures (35), (36), (37) and (38). These figures include the variations of performance parameters of a chemical rocket motor, theoretically calculated for Hydrogen-Oxygen-Nitrogen propellant system, with equilibrium flow approach; for combustion chamber pressures of 50, 100, 150 and 200 Psia and O/F ratios 2, 4, 6 and 8 at each chamber pressure.

Conclusions drawn from these figures are more or less similar to those mentioned in Section 3.2, Frozen Flow Approach. While comparing the results obtained from equilibrium flow approach with those obtained from the frozen flow approach, following can be concluded.

(1) At higher chamber pressures, performance parameters viz. specific impulse, characteristic velocity, equivalent exhaust belocity, linear exit velocity, average molecular weight, exit temperature etc. are having larger values than those theoretically calculated with the frozen flow approach. While both $\bar{\gamma}^*$ and thrust coefficient are bit lower.

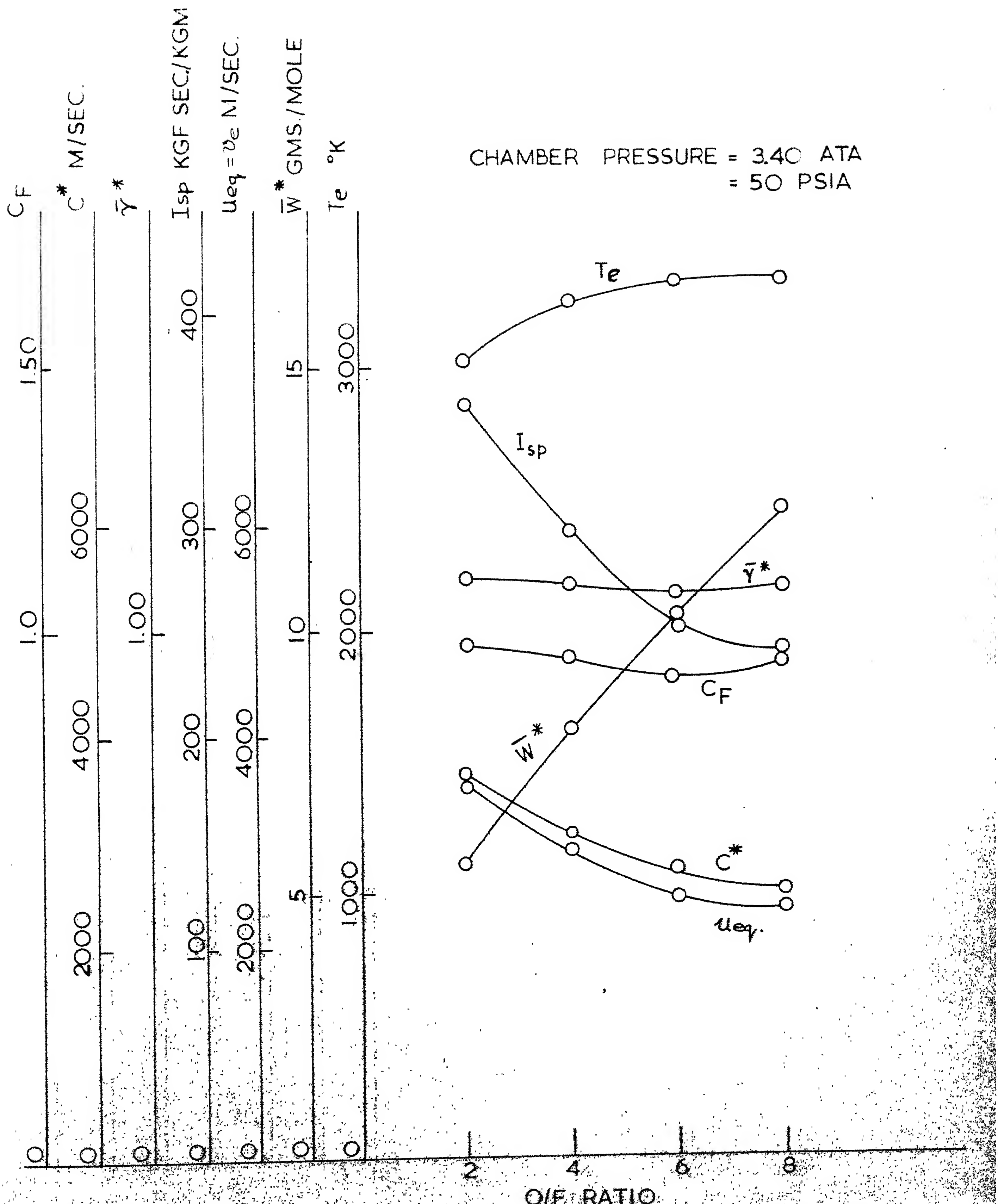


FIG.35_VARIATION OF THEORETICALLY CALCULATED PERFORMANCE PARAME FOR HYDROGEN - OXYGEN - NITROGEN PROPELLENT SYSTEM CHEMICAL ROCKET WITH O/F RATIOS ; WITH EQUILI. FLOW APPROACH

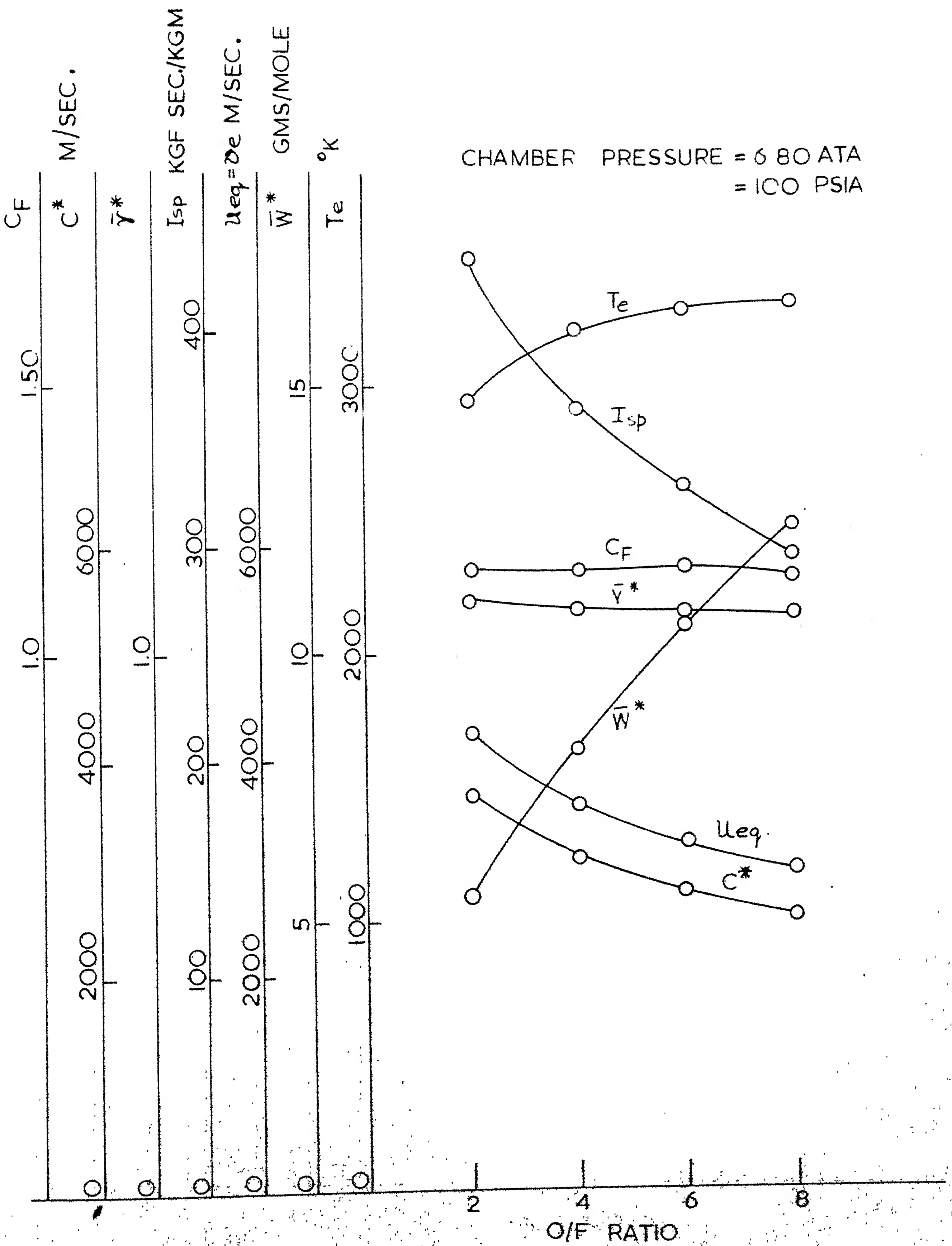


FIG 36_ VARIATION OF THEORETICALLY CALCULATED PERFORMANCE PARAMETERS FOR HYDROGEN - OXYGEN - NITROGEN PROPELLANT SYSTEM CHEMICAL ROCKET, WITH O/F RATIOS; WITH EQUILI. FLOW APPROACH

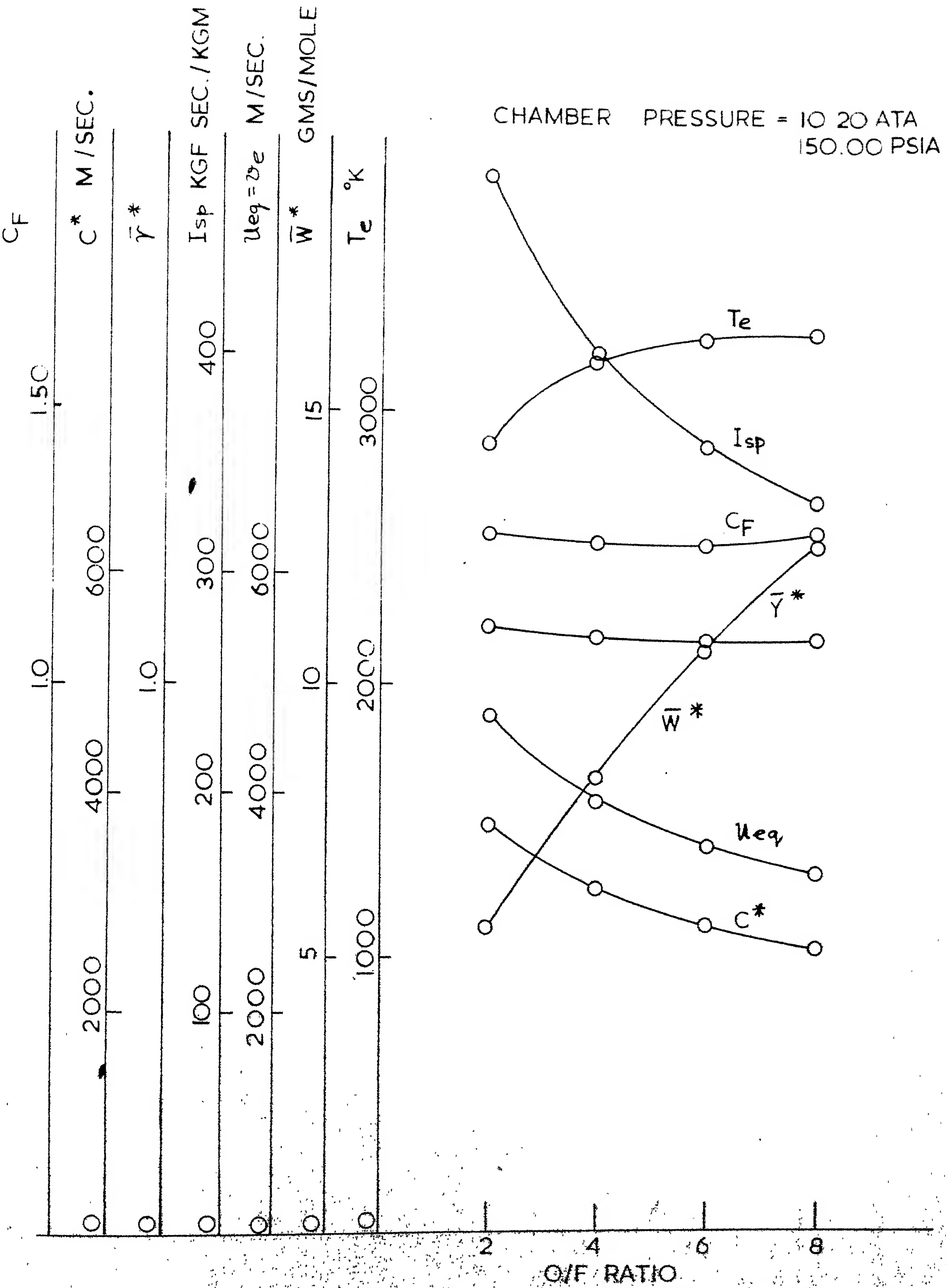


FIG.37_VARIATION OF THEORETICALLY CALCULATED PERFORMANCE PARAMETERS FOR HYDROGEN - OXYGEN - NITROGEN PROPELLANT SYSTEM CHEMICAL ROCKET, WITH O/F RATIOS, WITH EQUILI. FLOW APPROACH

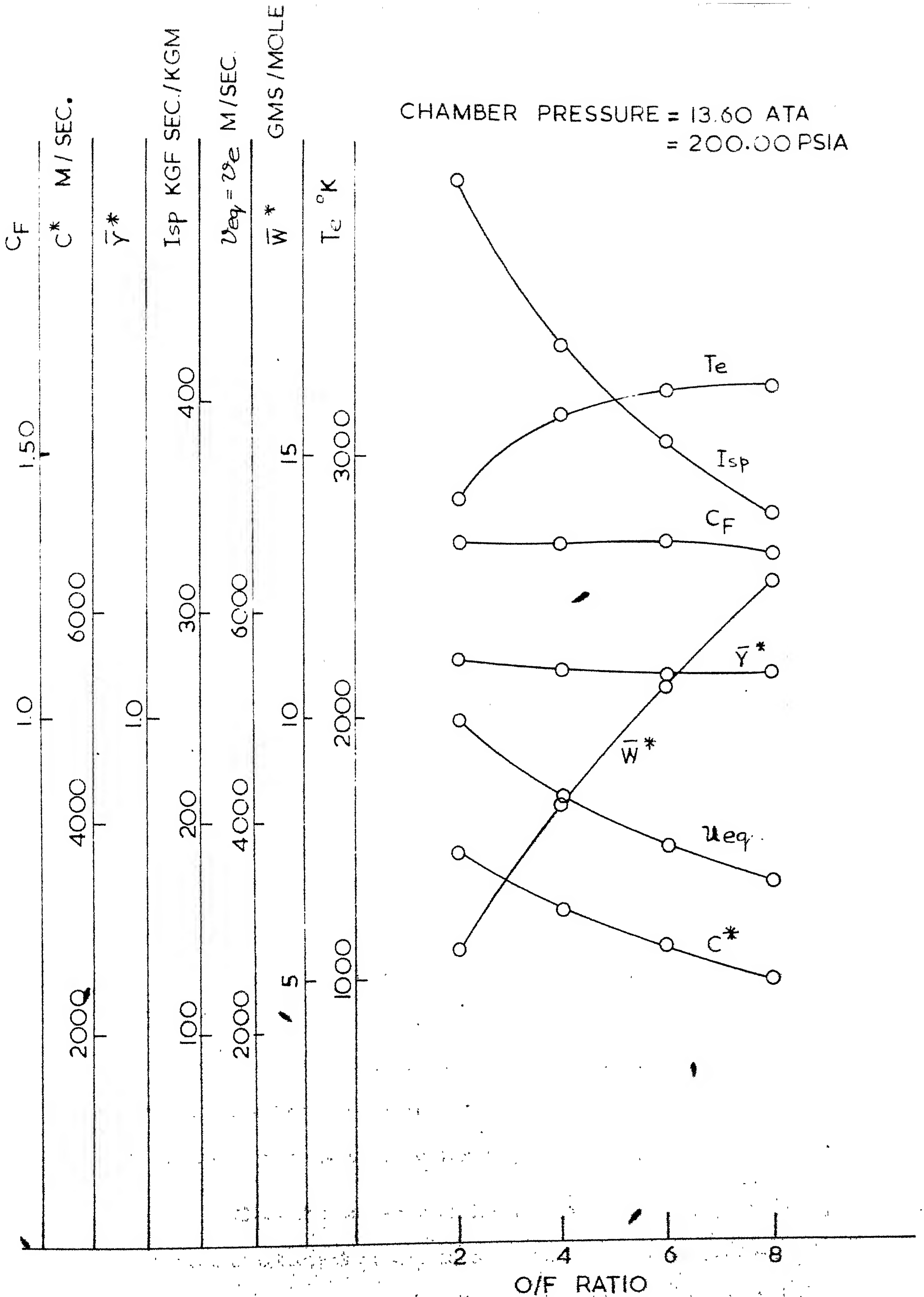


FIG.38_VARIATION OF THEORETICALLY CALCULATED PERFORMANCE PARAMETERS FOR HYDROGEN, OXYGEN, NITROGEN PROPELLANT SYSTEM CHEMICAL ROCKET, WITH O/F RATIOS; WITH EQUILI. FLOW APPROACH

(2) At lower chamber pressures, the theoretically calculated performance parameters like specific impulse and both equivalent exhaust velocity and linear exit velocity are lower than those by frozen flow approach. While exit temperature, average molecular weight, characteristic velocity etc. are larger for equilibrium flow approach case. Both $\bar{\gamma}^*$ and thrust coefficient are lower for equilibrium flow approach case. For very lower chamber pressure of 50 Psia, thrust coefficient has reduced to even less than 1.

Reason for conclusion No. (1) is the additional energy release, arises primarily from the recombination of dissociated and other species generated in combustion chamber which are not stable at lower temperatures. This is prominent with higher chamber pressures. While at lower chamber pressures because of favoured dissociation of species like $H_2O$, $H_2$, $O_2$, $N_2$ (at particular temperature) this additional energy release may be reduced and hence reducing the total heat drop in nozzle. Moreover because the average molecular weight also increases the ratio $\Delta H_c^e / \bar{W}^*$ reduces, reducing both linear exit velocity and equivalent exhaust velocity and hence specific impulse is also reduced. Increased characteristic velocities are due to lowered $\bar{\gamma}^*$, which overpowers molecular weight increase.

Also, for the case of lower chamber pressure, reduction of thrust coefficient is due to both reduction of equivalent exhaust velocity and increase of charateristic velocity. This

is prominent at chamber pressure of 50 Psia. Also note that $\bar{\gamma}^*$ has actually no meaning with changing composition flow i.e. equilibrium flow because this contradicts, one of the basic assumptions.

From Tables (53), (54), (55) and (56), following can be concluded as exceptional case, while making comparisions with frozen flow conclusions. For increasing combustion chamber pressure and at fixed O/F ratio, the average molecular weight here increases a bit.

## RECOMMENDATIONS

Experimental set-up can be extended to study

(1) Pressure distribution across the rocket nozzle.

(2) Heat transfer characteristics of water cooled rocket motor.

(3) Rocket flame spectroscopy and spectrometry.

Theoretical calculations can be extended with inclusion of condensed phases in product gas composition. Though theoretical propellant evaluation represents maximum performance and not necessarily that which can be achieved in practice, matching can be tried with different compositions of product gas by suppressing some of the species for each case.

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# APPENDIX A

## DATA, SAMPLE CALCULATIONS AND FINAL CALCULATED READINGS FOR EXPERIMENTAL APPROACH

A.1 Injector Flow Analysis

A.2 Flame-in Limits Establishment

A.3 Thrust Meter Analysis

A.4 Determination of Performance Parameters

## A.1 Injector Flow Analysis

Operational manual for LABROC VI provides injector flow calibration curve for SN/14 unit. This includes plot of flow rate in lb/sec. Vs. panel pressure in Psig for oxygen and methane gas, on which it was originally designed. The following data is given by Fig. (11).

TABLE 1

Injector Flow Calibration Data for Gaseous Propellant Methane.

| GAS USED | PENNEL PRESSURE IN Psig | FLOW RATE IN lbs/Sec. |
|---|---|---|
| Methane $CH_4$ With Molecular Weight 16 gms/mole | 82 | 0.00142 |
| | 170 | 0.00280 |
| | 270 | 0.00430 |

The mass flow rate of the propellant and the pressure drop across the injector are related by the equation

$$\dot{m} = C_d \rho A_{in} \sqrt{2 \Delta p/\rho} \tag{A.1.1}$$

For fixed $C_d$, $A_{in}$ and $\Delta p$

$$\dot{m} = \text{Const} \sqrt{\rho} = \text{Const} \sqrt{W} \tag{A.1.2}$$

For Hydrogen and Methane gas because we are selecting same pressures and $A_{in}$ and hence $C_d$ ( $\because C_d \sim \frac{1}{A_{in} \text{ Pressure Head}}$ )

$$\therefore \dot{m}_1 / \dot{m}_2 = \sqrt{W_1 / W_2} \tag{A.1.3}$$

Subscript 1 is for methane and 2 is for hydrogen. Also $W_1 = 16$ gms/mole and $W_2 = 2$ gms/mole.

$$\therefore \dot{m}_1 = \sqrt{8}\,\dot{m}_2 = 2.82843\,\dot{m}_2$$

$$\therefore \dot{m}_1 = 2.83\,\dot{m}_2 \qquad (A.1.4)$$

Thus at the same pressure settings we have

$$(\dot{m}_2)_1 = 0.00142/2.83 = 0.000502$$

$$(\dot{m}_2)_2 = 0.0028/2.83 = 0.000989$$

$$(\dot{m}_2)_3 = 0.0043/2.83 = 0.001519$$

Thus we have following data for injector flow calibration for Hydrogen gas.

TABLE 2

Injector Flow Calibration Data, Calculated For Gaseous Propellant Hydrogen.

| GAS USED | PANEL PRESSURE IN Psig | FLOW RATE IN Lb/Sec. |
|---|---|---|
| Hydrogen $H_2$ | 82 | 0.000502 |
| With Molecular | 170 | 0.000989 |
| Weight 2 Gms/Mole | 270 | 0.001519 |

Thus similar graph for $H_2$ calibration can be plotted. Same graph for $O_2$ is used see Fig. (12).

Now problem is to find out panel pressure settings for both $H_2$ and $O_2$ for various O/F ratio and at various mass flow rates through nozzle.

Take $\dot{m}$ = 0.009 lb/sec, through nozzle and O/F ratio of 3 is selected. Now

$$\text{Rate of } O_2 \text{ Flow} + \text{Rate of } H_2 \text{ Flow} = \dot{m} \qquad (A.1.5)$$

$\therefore$ 4 (Rate of $H_2$ Flow) = 0.009 lb/sec.

$\therefore$ Rate of $H_2$ Flow = 0.009/4 = 0.00225 and

Rate of $O_2$ Flow $\therefore$ 3 x 0.00225 i.e.

= 0.00675 both in lb/sec.

From Fig. (12) panel setting for $H_2$ gas is 205 Psig and for $O_2$ gas is 420 Psig. Same can be repeated for various O/F ratio and at various $\dot{m}$. Thus we have the following tables for calculated readings.

TABLE 3

Calculated Panel Pressure Settings for Gaseous Propellants Hydrogen and Oxygen at Different O/F ratios for $\dot{m}$ = 0.009 lb/sec.

| O/F RATIO | PANEL PRESSURE IN Psig FOR $O_2$ | PANEL PRESSURE IN Psig FOR $H_2$ |
|---|---|---|
| 3 | 205 | 420 |
| 4 | 220 | 350 |
| 5 | 230 | 270 |
| 6 | 237 | 225 |
| 7 | 242 | 195 |
| 8 | 246 | 170 |

TABLE 6

Calculated Panel Pressure Settings For Gaseous Propellants Hydrogen and Oxygen at Different O/F Ratios for $\dot{m}$ = 0.013 lb/sec.

| O/F RATIO | PANEL PRESSURE IN Psig FOR $O_2$ | PANEL PRESSURE IN Psig FOR $H_2$ |
|---|---|---|
| 3 | 315 | 560 |
| 4 | 328 | 495 |
| 5 | 342 | 405 |
| 6 | 350 | 345 |
| 7 | 357 | 295 |
| 8 | 395 | 283 |

Note : Actual sets of readings are restricted due to maximum range on regular being 600 Psig and due to leakage failure at high pressure regulator setting. Thus O/F ratio range is restricted.

A.2 Flame-in Limits Establishment

For rocket motor 202253 SN/14; Rocket Nozzle 202726, SN/4, $\epsilon$ 1 : 1, $D_t$ = 0.381 Cms.

TABLE 7

Observed Readings For Flame-in Limits Establishment Experiment For Fixed Oxygen Setting.

| RUN NO | 1 | 2 | 3 | 4 |
|---|---|---|---|---|
| FIXED SETTING FOR $O_2$ IN Psig | ~~100~~ | 200 | 300 | 400 |
| $O_2$ SETTING AT FIRE, IN Psig | 72 | 163 | 255 | 340 |
| $H_2$ SETTING AT FIRE, IN Psig | 30 | 43 | 50 | 60 |
| COMBUSTION CHAMBER PRESSURE AT FIRE, IN Psig | 00 | 05 | 10 | 16 |

TABLE 8

Observed Readings For Flame-in Limits Establishment Experiment For Fixed Hydrogen Setting.

| RUN NO | 1 | 2 | 3 | 4 |
|---|---|---|---|---|
| FIXED SETTING FOR $H_2$ IN Psig | 100 | 200 | 300 | 400 |
| $H_2$ SETTING AT FIRE, IN Psig | 90 | 185 | 280 | 375 |
| $O_2$ SETTING AT FIRE, IN Psig | 10 | 20 | 30 | 45 |
| COMBUSTION CHAMBER PRESSURE AT FIRE, IN Psig. | 00 | 00 | 00 | 00 |

Sample calculations are carried out for readings in Table 7. From Fig. (12) at pressures noted for $H_2$ and $O_2$, their flow rate is calculated.

At 72 Psig for $O_2$, flow rate is 0.00255 lb/sec.

At 30 Psig for $H_2$, flow rate is 0.00019 lb/sec

$\therefore$ O/F ratio = 0.00255/0.00019

= 13.40 also

Combustion chamber pressure is 14.7 Psia i.e. 1.033 $Kgf/cm^2$.

Calculated readings for data in Table (7) and Table (8) are as under.

TABLE 9

Calculated Data For Flame-in Limits Establishment Experiment For Fixed Oxygen Setting

| RUN NO | 1 | 2 | 3 | 4 |
|---|---|---|---|---|
| HIGH O/F RATIO | 13.40 | 20.00 | 25.20 | 28.80 |
| COMBUSTION CHAMBER PRESSURE AT FIRE, IN $Kgf/Cm^2$ | 1.033 | 1.34 | 1.68 | 2.09 |

TABLE 10

Calculated Data For Flame-in Limits Establishment Experiment For Fixed Hydrogen Setting

| RUN NO | 1 | 2 | 3 | 4 |
|---|---|---|---|---|
| LOW O/F RATIO | 0.788 | 0.755 | 0.740 | 0.730 |
| COMBUSTION CHAMBER PRESSURE AT FIRE, IN $Kgf/Cm^2$ | 1.033 | 1.033 | 1.033 | 1.033 |

For rocket motor 202253 SN/14, Rocket nozzle 202261, SN/21, $\epsilon$ 2.09 : 1, $D_t$ = 0.360 Cms.

TABLE 11

Observed Readings for Flame-in Limits Establishment Experiment For Fixed Oxygen Setting

| RUN NO | 1 | 2 | 3 | 4 |
|---|---|---|---|---|
| FIXED SETTING FOR $O_2$ IN Psig | 100 | 200 | 300 | 400 |
| $O_2$ SETTING AT FIRE, IN Psig | 80 | 175 | 265 | 360 |
| $H_2$ SETTING AT FIRE, IN Psig | 30 | 100 | 160 | 225 |
| COMBUSTION CHAMBER PRESSURE AT FIRE IN Psig | 00 | 00 | 40 | 62 |

TABLE 12

Observed Readings for Flame-in Limits Establishment Experiment For Fixed Hydrogen Setting.

| RUN NO | 1 | 2 | 3 | 4 |
|---|---|---|---|---|
| FIXED SETTING FOR $H_2$ IN Psig | 100 | 200 | 300 | 400 |
| $H_2$ SETTING AT FIRE, IN Psig | 85 | 175 | 265 | 350 |
| $O_2$ SETTING AT FIRE, IN Psig | 10 | 25 | 50 | 70 |
| COMBUSTION CHAMBER PRESSURE AT FIRE, IN Psig | 00 | 00 | 00 | 00 |

Calculated readings for data in Table (11) and Table (12) are as under.

TABLE 13

Calculated Data For Flame-in Limits Establishment Experiment For Fixed Oxygen Setting.

| RUN NO | 1 | 2 | 3 | 4 |
|---|---|---|---|---|
| HIGH O/F RATIO | 14.75 | 9.675 | 8.825 | 8.59 |
| COMBUSTION CHAMBER PRESSURE IN $Kgf/Cm^2$ | 1.033 | 1.033 | 3.72 | 5.225 |

TABLE 14

Calculated Data For Flame-in Limited Establishment Experiment For Fixed Hydrogen Setting.

| RUN NO | 1 | 2 | 3 | 4 |
|---|---|---|---|---|
| LOW O/F RATIO | 0.83 | 0.956 | 1.30 | 1.317 |
| COMBUSTION CHAMBER PRESSURE AT FIRE, IN $Kgf/Cm^2$ | 1.033 | 1.033 | 1.033 | 1.033 |

A.3 Thrust Meter Analysis

From the set-up as shown in Fig. 10 it is clear that how one should calibrate the thrust meter before taking a set of performance parameters measurement.

For rocket motor 202253, SN/14; Rocket nozzle 202726

SN/4, $\epsilon$ 1 : 1, $D_t$ = 0.381 Cms.

TABLE 15

Observed Readings For Thrust Meter Calibration Experiment For Performance Analysis Set 1.

| LOADS IN Lbs | STRAIN IN $\mu$In/In + ve |
|---|---|
| 2.11 | 4400 |
| 3.11 | 4526 |
| 4.11 | 4676 |
| 5.22 | 4830 |

TABLE 16

Observed Readings For Thrust Meter Calibration Experiment For Performance Analysis Set 2.

| LOADS IN Lbs | STRAIN IN $\mu$In/In + ve |
|---|---|
| 2.11 | 4554 |
| 3.11 | 4696 |
| 4.11 | 4832 |
| 5.22 | 4970 |

TABLE 17

Observed Readings For Thrust Meter Calibration Experiment For Performance Analysis Set 3.

| LOAD IN Lbs | STRAIN IN $\mu$In/In + ve |
|---|---|
| 2.11 | 4500 |
| 3.11 | 4628 |
| 4.11 | 4766 |
| 5.22 | 4908 |

For rocket motor 202253, SN/14; Rocket nozzle 202261, SN/21, $\epsilon$ 2.09 : 1, $D_t$ 0.360 Cms.

TABLE 18

Observed Readings For Thrust Meter Calibration Experiment For Performance Analysis Set 4.

| LOAD IN Lbs | STRAIN IN $\mu$In/In + ve |
|---|---|
| 2.11 | 4740 |
| 3.11 | 4870 |
| 4.11 | 4992 |
| 5.22 | 5114 |

TABLE 19

Observed Readings For Thrust Meter Calibration Experiment For Performance Analysis Set 5.

| LOAD IN Lbs. | STRAIN IN $\mu$In/In + ve |
|---|---|
| 2.11 | 4760 |
| 3.11 | 4877 |
| 4.11 | 4975 |
| 5.22 | 5094 |

TABLE 20

Observed Readings For Thrust Meter Calibration Experiment For Performance Analysis Set 6.

| LOAD IN Lbs. | STRAIN IN $\mu$In/In + ve |
|---|---|
| 2.11 | 4770 |
| 3.11 | 4864 |
| 4.11 | 4982 |
| 5.22 | 5088 |

All thrust meter calibration readings are taken with cooling water circulation on.

A.4 Determination of Performance Parameters.

For rocket motor 202253, SN/14; Rocket nozzle 202726 SN/4, $\epsilon$ 1 : 1, $D_t$ = 0.381 Cms.

TABLE 21

Observed Readings For Experiment of Determination of Performance Parameters of A Gaseous Rocket Motor Using Hydrogen and Oxygen At Different O/F Ratios For $\dot{m}$ = 0.009 lb/sec.

Performance Analysis Set 1

| O/F RATIO | $P_c$ Psig | TMR* $\mu$In/In + ve | COOLANT FLOW RATE Lbs/Min. | COOLANT INLET TEMP. °F | COOLANT OUTLET TEMP. °F |
|---|---|---|---|---|---|
| 8 | 32 | 4560 | 15.20 | 68 | 84 |
| 7 | 35 | 4570 | 15.00 | 68 | 84 |
| 6 | 40 | 4588 | 15.40 | 68 | 88 |
| 5 | 46 | 4600 | 15.20 | 68 | 96 |
| 4 | 58 | 4620 | 15.40 | 68 | 108 |

* Thrust meter readings with initial load = 2.11 Lbs.

For this case sample calculations are as under.

O/F ratio is selected as 8 for sample calculation and all other data corresponding to it is taken.

Combustion chamber pressure in Kgf/Cm$^2$

$$= (\text{Observed reading} + 14.7) / 14.7$$

$$= (32 + 14.7) / 14.7 = 3.18 \text{ Kgf/Cm}^2$$

Now thrust calculations are to be done.

Thrust meter reading is 4560 $\mu$ In/In + ve. See Fig. (15) and read correspondingly the load of 3.32 Lbs. Actual additional force = Thrust is 3.32 - 2.11 = 1.21 Lbs.

$$\text{Thrust} \quad F = 1.21/2.2 = 0.551 \text{ Kgf.}$$

$$\text{Specific Impulse} = F/\dot{m} = \frac{1.21 \times 2.2}{0.009 \times 2.2} = 134.5 \text{ Kgf Sec./Kgm}$$

$$\text{Thrust Coefficient} = F/p_c\ A_t = \frac{1.21 \times 6.42}{46.7 \times 0.114} \quad \text{OR}$$

$$= \frac{0.551}{0.114 \times 3.18} = 1.48$$

where $A_t = \pi/4\ D_t^2$ where $D_t = 0.381$ Cms.

$$\text{Characteristic Velocity} = p_c\ A_t / \dot{m}$$

$$= \frac{32.2 \times 46.7 \times 0.114}{0.009 \times 6.42}$$

$$= 3000 \text{ Ft/Sec.} \quad \text{OR}$$

$$= \frac{3.18 \times 0.114 \times 9.81}{0.009 / 2.2}$$

$$= 915 \text{ M/Sec.}$$

$$\text{Equivalent Exhaust Velocity} = \text{Specific Impulse} \times g$$

$$= 134.5 \times 9.81$$

$$= 1322 \text{ M/Sec.}$$

To get more correct results, carry out all calculations in FPS System and finally convert them in MKS System. Though both systems are used here. For data in Table (21) calculated readings are tabulated as under.

TABLE 22

Calculated Data For Experiment of Determination Of Performance Parameters Of A Gaseous Rocket Motor Using Hydrogen and Oxygen, At Different O/F Ratios For $\dot{m}$ = 0.009 Lb/Sec.

| O/F RATIO | $p_c$ Kgf/Cm$^2$ | F Kgf. | $I_{sp}$ Kgf Sec/Kgm | $C_F$ | C* M/Sec. | $u_{eq}$ M/Sec. |
|---|---|---|---|---|---|---|
| 8 | 3.18 | 0.551 | 134.50 | 1.48 | 915 | 1322 |
| 7 | 3.38 | 0.586 | 143.50 | 1.465 | 970 | 1415 |
| 6 | 3.72 | 0.636 | 155.70 | 1.440 | 1060 | 1540 |
| 5 | 4.13 | 0.678 | 165.50 | 1.380 | 1182 | 1632 |
| 4 | 4.94 | 0.745 | 182.10 | 1.270 | 1420 | 1800 |

TABLE 23

Observed Readings For Experiment of Determination of Performance Parameters of A Gaseous Rocket Motor Using Hydrogen and Oxygen At Different O/F Ratios For $\dot{m}$ = 0.011 Lb/Sec.

Performance Analysis Set 2

| O/F RATIO | $p_c$ Psig | TMR* $\mu$In/In + ve | COOLANT FLOW RATE Lbs/Min. | COOLANT INLET TEMP °F | COOLANT OUTLET TEMP. °F |
|---|---|---|---|---|---|
| 8 | 50 | 4796 | 15.2 | 64 | 82 |
| 7 | 56 | 4798 | 15.2 | 64 | 84 |
| 6 | 61 | 4802 | 15.2 | 64 | 86 |
| 5 | 69 | 4805 | 15.3 | 64 | 90 |
| 4 | 79 | 4806 | 15.2 | 64 | 98 |

*Thrust meter readings with initial load = 2.11 Lbs.

Use Fig. (16) for calculations. Calculated readings for the data in Table (23) are as under.

TABLE 24

Calculated Data For Experiment of Determination And Of Performance Parameters Of A Gaseous Rocket Motor Using Hydrogen and Oxygen, At Different O/F Ratios For $\dot{m}$ = 0.011 Lb/Sec.

| O/F RATIO | $p_c$ Kgf/Cm$^2$ | F Kgf. | $I_{sp}$ Kgf Sec/Kgm | $C_F$ | C* M/Sec. | $u_{eq}$ M/Sec. |
|---|---|---|---|---|---|---|
| 8 | 4.49 | 0.792 | 158.1 | 1.51 | 1035 | 1551 |
| 7 | 4.92 | 0.800 | 160.0 | 1.40 | 1102 | 1570 |
| 6 | 5.27 | 0.809 | 161.75 | 1.33 | 1212 | 1585 |
| 5 | 5.82 | 0.816 | 163.00 | 1.21 | 1338 | 1594 |
| 4 | 6.52 | 0.821 | 164.00 | 1.05 | 1500 | 1609 |

TABLE 25

Observed Readings For Experiment of Determination of Performance Parameters of A Gaseous Rocket Motor Using Hydrogen and Oxygen At Different O/F Ratios For $\dot{m}$ = 0.013 Lb/Sec.

Performance Analysis Set 3

| O/F RATIO | $p_c$ Psig | TMR* $\mu$In/In + ve | COOLANT FLOW RATE Lbs/Min. | COOLANT INLET TEMP °F | COOLANT OUTLET TEMP. °F |
|---|---|---|---|---|---|
| 8 | 72 | 4727 | 15.2 | 64 | 80 |
| 7 | 73 | 4730 | 15.2 | 64 | 84 |
| 6 | 78 | 4740 | 15.2 | 64 | 88 |
| 5 | 87 | 4744 | 15.2 | 64 | 98 |

* Thrust meter readings with initial load = 2.11 Lbs.

Use Fig. (17) for calculations. Calculated readings for data in Table (25) are as under.

TABLE 26

Calculated Data For Experiment of Determination And Of Performance Parameters Of A Gaseous Rocket Motor Using Hydrogen and Oxygen, At Different O/F Ratios For $\dot{m}$ = 0.013/Lb/Sec.

| O/F RATIO | $p_c$ Kgf/Cm$^2$ | F Kgf. | $I_{sp}$ Kgf.Sec/Kgm | $C_F$ | C* M/Sec. | $u_{eq}$ M/Sec. |
|---|---|---|---|---|---|---|
| 8 | 5.89 | 0.782 | 132.10 | 1.120 | 1172 | 1299 |
| 7 | 5.96 | 0.793 | 134.50 | 1.117 | 1185 | 1320 |
| 6 | 6.31 | 0.836 | 141.75 | 1.115 | 1253 | 1390 |
| 5 | 6.94 | 0.858 | 145.50 | 1.048 | 1372 | 1430 |

For rocket motor 202253, SN/14; Rocket nozzle 202261, SN/21, $\epsilon$ 2.09 : 1, $D_t$ = 0.360 Cms

TABLE 27

Observed Readings For Experiment of Determination of Performance Parameters of A Gaseous Rocket Motor Using Hydrogen and Oxygen At Different O/F Ratios For $\dot{m}$ = 0.009 Lb/Sec.

Performance Analysis Set 4

| O/F RATIO | $p_c$ Psig | TMR* $\mu$ In/In + ve | COOLANT FLOW RATE Lbs/Min. | COOLANT INLET TEMP °F | COOLANT OUTLET TEMP. °F |
|---|---|---|---|---|---|
| 8 | 40 | 4894 | 14.6 | 66 | 82 |
| 7 | 43 | 4906 | 14.5 | 66 | 86 |
| 6 | 48 | 4910 | 14.6 | 66 | 88 |
| 5 | 56 | 4916 | 14.8 | 66 | 96 |
| 4 | 66 | 4930 | 14.8 | 66 | 104 |

* Thrust meter readings with initial load = 2.11 Lbs

Use Fig. (18) for calculations. Calculated readings for data in Table (27) are as under.

TABLE 28

Calculated Data For Experiment of Determination And Of Performance Parameters Of A Gaseous Rocket Motor Using Hydrogen and Oxygen, At Different O/F Ratios For $\dot{m}$ = 0.009 Lb/Sec.

| O/F RATIO | $p_c$ Kgf/Cm$^2$ | F Kgf. | $I_{sp}$ Kgf.Sec/Kgm | $C_F$ | C* M/Sec. | $u_{eq}$ M/Sec. |
|---|---|---|---|---|---|---|
| 8 | 3.72 | 0.5515 | 136 | 1.40 | 947 | 1335 |
| 7 | 3.92 | 0.605 | 148 | 1.45 | 1000 | 1452 |
| 6 | 4.27 | 0.623 | 152.5 | 1.37 | 1086 | 1495 |
| 5 | 4.82 | 0.636 | 156 | 1.245 | 1225 | 1530 |
| 4 | 5.50 | 0.723 | 168 | 1.176 | 1400 | 1640 |

TABLE 29

Observed Readings For Experiment Of Determination of Performance Parameters of A Gaseous Rocket Motor Using Hydrogen and Oxygen At Different O/F Ratios For $\dot{m}$ = 0.011 Lb/Sec.

Performance Analysis Set 5

| O/F RATIO | $p_c$ Psig | TMR* $\mu$In/In + ve | COOLANT FLOW RATE Lbs/Min. | COOLANT INLET TEMP °F | COOLANT OUTLET TEMP. °F |
|---|---|---|---|---|---|
| 8 | 54 | 4928 | 15 | 67 | 84 |
| 7 | 65 | 4934 | 15 | 67 | 90 |
| 6 | 74 | 4945 | 15.2 | 67 | 92 |
| 5 | 80 | 4952 | 15.2 | 67 | 94 |

*Thrust meter readings with initial load = 2.11 Lbs.

Use Figure (19) for calculations. Calculated readings for data in Table (29) are as under.

TABLE 30

Calculated Data For Experiment of Determination And Of Performance Parameters Of A Gaseous Rocket Motor Using Hydrogen and Oxygen, At Different O/F Ratios For $\dot{m}$ = 0.011 Lb/Sec.

| O/F RATIO | $p_c$ Kgf/Cm$^2$ | F Kgf. | $I_{sp}$ Kgf.Sec/Kgm | $C_F$ | C* M/Sec. | $u_{eq}$ M/Sec. |
|---|---|---|---|---|---|---|
| 8 | 4.68 | 0.708 | 141.9 | 1.425 | 976 | 1390 |
| 7 | 5.43 | 0.731 | 146.2 | 1.250 | 1132 | 1432 |
| 6 | 5.96 | 0.776 | 156.3 | 1.215 | 1262 | 1532 |
| 5 | 6.45 | 0.804 | 161 | 1.175 | 1349 | 1585 |

TABLE 31

Observed Readings For Experiment of Determination of Performance Parameters of A Gaseous Rocket Motor Using Hydrogen and Oxygen At Different O/F Ratios For $\dot{m}$ = 0.012 Lb/Sec.

Performance Analysis Set 6

| O/F RATIO | $p_c$ Psig | TMR* $\mu$In/In +ve | COOLANT FLOW RATE Lbs/Min. | COOLANT INLET TEMP °F | COOLANT OUTLET TEMP. °F |
|---|---|---|---|---|---|
| 8 | 63 | 4944 | 14.9 | 66 | 86 |
| 7 | 68 | 4946 | 14.8 | 66 | 88 |
| 6 | 76 | 4948 | 14.8 | 66 | 98 |
| 5 | 84 | 4952 | 14.8 | 66 | 100 |

* Thrust meter readings with initial load = 2.11 Lbs.

Use Fig. (20) for calculations. Calculated readings for data in Table (31) are as under.

TABLE 32

Calculated Data For Experiment of Determination And Of Performance Parameters Of A Gaseous Rocket Motor Using Hydrogen and Oxygen, At Different O/F Ratios For $\dot{m}$ = 0.012 Lb/Sec.

| O/F RATIO | $p_c$ Kgf/Cm$^2$ | F Kgf. | $I_{sp}$ Kgf Sec/Kgm | $C_F$ | C* M/Sec. | $u_{eq}$ M/Sec. |
|---|---|---|---|---|---|---|
| 8 | 5.28 | 0.766 | 141 | 1.365 | 1010 | 1385 |
| 7 | 5.63 | 0.776 | 142.5 | 1.30 | 1075 | 1399 |
| 6 | 6.175 | 0.790 | 145 | 1.208 | 1180 | 1421 |
| 5 | 6.72 | 0.808 | 148.3 | 1.136 | 1282 | 1455 |

# APPENDIX B

## DATA, COMPUTER PROGRAMME AND CALCULATED READINGS FOR THEORETICAL APPROACH

B.1 Determination Of Combustion Temperature and Composition of Product Gas

B.2 Frozen Flow Approach

B.3 Equilibrium Flow Approach

## B.1 Determination Of Combustion Temperature and Composition Of Product Gas

Numerical solution of the equation (2.1.27) Section II is to be obtained. To solve the set, equations are so arranged that the diagonal elements of the resultant matrix are non zero i.e.

$$
\begin{bmatrix}
n_1 & n_2 & n_3 & n_4 & n_5 & n_6 & n_7 & n_8 & n_9 & 0 & 0 \\
n_1 & n_2 & n_3 & 0 & 2n_5 & 0 & 0 & n_8 & 0 & (-bA)_{Calc} & 0 \\
0 & 0 & n_3 & 0 & 0 & 2n_6 & 0 & 0 & n_9 & (-cA)_{Calc} & 0 \\
-1 & 1 & 0 & 1/2 & 0 & 0 & 0 & 0 & 0 & 0 & 0 \\
1 & 0 & 0 & -1 & -1/2 & 0 & 0 & 0 & 0 & 0 & 0 \\
0 & 0 & 1 & 0 & -1/2 & -1/2 & 0 & 0 & 0 & 0 & 0 \\
0 & 0 & 0 & -1/2 & 0 & 0 & 1 & 0 & 0 & 0 & 0 \\
0 & 0 & 0 & 0 & -1/2 & 0 & 0 & 1 & 0 & 0 & 0 \\
0 & 0 & 0 & 0 & 0 & -1/2 & 0 & 0 & 1 & 0 & 0 \\
2n_1 & n_2 & 0 & 2n_4 & 0 & 0 & n_7 & 0 & 0 & (-aA)_{Calc} & 0 \\
H^T_{o_1} n_1 & H^T_{o_2} n_2 & H^T_{o_3} n_3 & H^T_{o_4} n_4 & H^T_{o_5} n_5 & H^T_{o_6} n_6 & H^T_{o_7} n_7 & H^T_{o_8} n_8 & H^T_{o_9} n_9 & (-hA)_{Calc} & (\sum_i C^o_{p_i} n_i) T
\end{bmatrix}
\begin{bmatrix}
\Delta \ln n_1 \\ \Delta \ln n_2 \\ \Delta \ln n_3 \\ \Delta \ln n_4 \\ \Delta \ln n_5 \\ \Delta \ln n_6 \\ \Delta \ln n_7 \\ \Delta \ln n_8 \\ \Delta \ln n_9 \\ \Delta \ln A \\ \Delta \ln T
\end{bmatrix}
=
\begin{bmatrix}
P_{Calc} \ln\left(\frac{P_{True}}{P_{Calc}}\right) \\
(bA)_{Calc} \ln\left(\frac{b_{True}}{b_{Calc}}\right) \\
(CA)_{Calc} \ln\left(\frac{C_{True}}{C_{Calc}}\right) \\
\ln\left(\frac{R_{2\,True}}{R_{2\,Calc}}\right) \\
\ln\left(\frac{R_{1\,True}}{R_{1\,Calc}}\right) \\
\ln\left(\frac{R_{3\,True}}{R_{3\,calc}}\right) \\
\ln\left(\frac{R_{4\,True}}{R_{4\,Calc}}\right) \\
\ln\left(\frac{R_{5\,True}}{R_{5\,calc}}\right) \\
\ln\left(\frac{R_{6\,True}}{R_{6\,Calc}}\right) \\
(aA)_{Calc} \ln\left(\frac{a_{True}}{a_{calc}}\right) \\
(hA)_{calc} \ln\left(\frac{h_{True}}{h_{calc}}\right)
\end{bmatrix}
\tag{B.1.1}
$$

is the final set to be solved.

Solution of set (B.1.1) is obtained using the standard disc programme available at I.I.T. Kanpur, Computer Center Library. This solves the given set using Gauss - Jordon method. Following steps were carefully followed to achieve convergence for the solution.

(1) Sum of the initial guess of $n_i$ should be preferably less than $p_c$ (this is preferable).

(2) The estimated temperature is held constant for several iterations, thereby allowing adjustment of partial pressures to some more reasonable values.

(3) Variation of temperature then by steps from starting value of 5000 °K till some reasonable error limit is established between initial guess of temperature and calculated value of temperature.

(4) Oscillations in solution can be avoided by modifying the error limits mentioned in (3).

Equilibrium temperature and composition is to be determined at pressures of 50, 100, 150 and 200 Psia in combustion chamber for OFR varying from 2, 4, 6 and 8. All necessary thermodynamic data for all species are taken from standard tables given in References (19) and (29). A suitable temperature interval of 200 °K is chosen. Attached herewith is the developed Computer programme for system IBM 7044/1401, which will provide the method of solution.

```
CIBJOB          MAP
CIBFTC MAIN
C      COMBUSTION TEMP.AND COMPOSITION DETERMINATION FOR GIVEN O/F RATIO
C      AND COMBUSTION PRESSURE FOR H2-O2 GAS ROCKET MOTOR.
C      READING OF VARIUS VALUES MENTIONED IN DIMENSION STATEMENT FROM
C      STANDARD TABLES AVAILABLE
C      UNITS OF ENTHALPY ARE KCAL/MOLE.
C      UNITS OF SP HEATS ARE CAL/OK-MOLE.
       DIMENSION AK1(24),AK2(24),AK3(24),AK4(24),AK5(24),AK6(24),
      1HOT1(24),HOT2(24),HOT3(24),HOT4(24),HOT5(24),HOT6(24),HOT7(24),
      2HOT8(24),HOT9(24),CPT1(24),CPT2(24),CPT3(24),CPT4(24),CPT5(24),
      3CPT6(24),CPT7(24),CPT8(24),CPT9(24),T(24),OFR(4),A(11,12),B(11,1),
      4X(11),P(11),XJ(9)
       REAL N(9),NJ(9),NT
       DATA BTRUE/2. /
       READ 2,L
     2 FORMAT (I2)
       READ 3,M
     3 FORMAT (I1)
       READ 4,(T(I),I=1,24)
     4 FORMAT  (8F10.1)
       READ 6,(AK1(I),I=1,24)
     6 FORMAT (6(E10.4,2X))
       READ 6,(AK2(I),I=1,24)
       READ 8,(AK3(I),I=1,24)
     8 FORMAT (6(E11.5,1X))
       READ 8,(AK4(I),I=1,24)
       READ 8,(AK5(I),I=1,24)
       READ 10,(AK6(I),I=1,24)
    10 FORMAT (4(E14.8,1X))
       READ 12,(HOT1(I),I=1,24)
    12 FORMAT (8F10.4)
       READ 12,(HOT2(I),I=1,24)
       READ 12,(HOT3(I),I=1,24)
       READ 12,(HOT4(I),I=1,24)
       READ 12,(HOT5(I),I=1,24)
       READ 12,(HOT6(I),I=1,24)
       READ 12,(HOT7(I),I=1,24)
       READ 12,(HOT8(I),I=1,24)
       READ 12,(HOT9(I),I=1,24)
       READ 14,(CPT1(I),I=1,24)
       READ 14,(CPT2(I),I=1,24)
       READ 14,(CPT3(I),I=1,24)
       READ 16,(CPT4(I),I=1,24)
    14 FORMAT (8F10.3)
    16 FORMAT (8F10.4)
       READ 14,(CPT5(I),I=1,24)
       READ 16,(CPT6(I),I=1,24)
       READ 14,(CPT7(I),I=1,24)
       READ 16,(CPT8(I),I=1,24)
```

```
      READ 16,(CPT9(I),I=1,24)
      READ 18,(OFR(JJ),JJ=1,4)
   18 FORMAT (4F1 .1)
      DO 1111 JJ=1,4
      CALL FLUN(1    )
      INDEX=
      K=
C     PREPARATIONS FOR MAKING CURRENT VARIABLES.
      DO 853 I=1,9
  853 N(I)= .25
      ANEQW=1.0
      CPTRA=3.4
C     STARTING TEMPARATURE IS ASSUMED TO BE 5    OKELVIN.
      AT=T(24)
      AAK1=AK1(24)
      AAK2=AK2(24)
      AAK3=AK3(24)
      AAK4=AK4(24)
      AAK5=AK5(24)
      AAK6=AK6(24)
      AHOT1=HOT1(24)
      AHOT2=HOT2(24)
      AHOT3=HOT3(24)
      AHOT4=HOT4(24)
      AHOT5=HOT5(24)
      AHOT6=HOT6(24)
      AHOT7=HOT7(24)
      AHOT8=HOT8(24)
      AHOT9=HOT9(24)
      ACPT1=CPT1(24)
      ACPT2=CPT2(24)
      ACPT3=CPT3(24)
      ACPT4=CPT4(24)
      ACPT5=CPT5(24)
      ACPT6=CPT6(24)
      ACPT7=CPT7(24)
      ACPT8=CPT8(24)
      ACPT9=CPT9(24)
C     FORMULATION OF CURRENT VARIABLES ENDS HERE.
C     CALCULATION FOR VARIOUS ELEMENTS OF MATRIX A(I,J) IS STARTED.
   68 SUMN=N(1)+N(2)+N(3)+N(4)+N(5)+N(6)+N(7)+N(8)+N(9)
      AR1TR=AAK1
      AR2TR=AAK2
      AR3TR=AAK3
      AR4TR=AAK4
      AR5TR=AAK5
      AR6TR=AAK6
      ATRUE=(16.00/OFR(JJ))*2.
      FACOIM=((16.00/OFR(JJ))+1.00)*0.09
      CTRUE=FACOIM*2.0
```

```
      AACALC=2.*N(1)+N(2)+2.*N(4)+N(7)
      BACALC=N(1)+N(2)+N(3)+2.*N(5)+N(8)
      CACALC=N(3)+2.*N(6)+N(9)
      ACALC=AACALC/ANEQW
      BCALC=BACALC/ANEQW
      CCALC=CACALC/ANEQW
      PTRA=CPTRA
      PCALC=SUMN
      AR1CAL=(N(1))/(N(4)*(N(5)**0.5))
      AR2CAL=((N(2))*(N(4)**(0.5)))/(N(1))
      AR3CAL=(N(3))/((N(6)**0.5)*(N(5)**0.5))
      AR4CAL=(N(7))/(N(4)**0.5)
      AR5CAL=(N(8))/(N(5)**0.5)
      AR6CAL=(N(9))/(N(6)**0.5)
      HACALC=AHOT1*N(1)+AHOT2*N(2)+AHOT3*N(3)+AHOT4*N(4)+AHOT5*N(5)+
     1AHOT6*N(6)+AHOT7*N(7)+AHOT8*N(8)+AHOT9*N(9)
      HCALC=HACALC/ANEQW
      HTRUE=(16.0 /OFR(JJ))*(AHOT4)+AHOT5+FACOIM*AHOT6
      SCPNT=ACPT1*N(1)+ACPT2*N(2)+ACPT3*N(3)+ACPT4*N(4)+ACPT5*N(5)+
     1ACPT6*N(6)+ACPT7*N(7)+ACPT8*N(8)+ACPT9*N(9)
C     PREREQUISITE CALCULATIONS FOR ELEMENTS OF MATRIX ENDS HERE.
C      ACTUAL CALCULATIONS OF THE ELEMENTS OF MATRIX STARTS.
      DO 1500 LL=1,11
      DO 1500 MM=1,12
      A(LL,MM)=0.00
 1500 CONTINUE
      A(1,1)=N(1)
      A(1,2)=N(2)
      A(1,3)=N(3)
      A(1,4)=N(4)
      A(1,5)=N(5)
      A(1,6)=N(6)
      A(1,7)=N(7)
      A(1,8)=N(8)
      A(1,9)=N(9)
      A(1,12)=(PCALC)*ALOG(PTRA/PCALC)
      A(2,1)=N(1)
      A(2,2)=N(2)
      A(2,3)=N(3)
      A(2,5)=2.*N(5)
      A(2,8)=N(8)
      A(2,10)=-(BACALC)
      A(2,12)=(BACALC)*ALOG(BTRUE/BCALC)
      A(3,3)=N(3)
      A(3,6)=2.*N(6)
      A(3,9)=N(9)
      A(3,10)=-(CACALC)
      A(3,12)=(CACALC)*ALOG(CTRUE/CCALC)
      A(4,1)=-1.00
      A(4,2)=1.00
```

```
      A(4,4)=+0.5
      A(4,12)=ALOG(AR2TR/AR2CAL)
      A(5,1)=1.
      A(5,4)=-1.0
      A(5,5)=-0.5
      A(5,12)=ALOG(AR1TR/AR1CAL)
      A(6,3)=1.
      A(6,5)=-0.5
      A(6,6)=-0.5
      A(6,12)=ALOG(AR3TR/AR3CAL)
      A(7,4)=-0.5
      A(7,7)=1.00
      A(7,12)=ALOG(AR4TR/AR4CAL)
      A(8,5)=-0.5
      A(8,8)=1.0
      A(8,12)=ALOG(AR5TR/AR5CAL)
      A(9,6)=-0.5
      A(9,9)=1.0
      A(9,12)=ALOG(AR6TR/AR6CAL)
      A(10,1)=2.*N(1)
      A(10,2)=N(2)
      A(10,4)=2.*N(4)
      A(10,7)=N(7)
      A(10,10)=-(AACALC)
      A(10,12)=(AACALC)*ALOG(ATRUE/ACALC)
      IF (INDEX.EQ.1) GO TO 45
      DO 47 I=1,10
   47 B(I,1)=A(I,12)
      CALL MATINV(A,10,B,1,DET)
      DO 49 J=1,10
   49 X(J)=B(J,1)
      DO 880 I=1,9
      P(I)=X(I)+ALOG(N(I))
      N(I)=EXP(P(I))
  880 CONTINUE
      P(10)=X(10)+ALOG(ANEQW)
      ANEQW=EXP(P(10))
      SNPPA=N(1)+N(2)+N(3)+N(4)+N(5)+N(6)+N(7)+N(8)+N(9)
      PERROR=((SNPPA-CPTRA)/(CPTRA))*100.0
      IF ((ABS(PERROR)).LE.5.)    INDEX=1
      GO TO 68
   45 A(11,1)=AHOT1*N(1)
      A(11,2)=AHOT2*N(2)
      A(11,3)=AHOT3*N(3)
      A(11,4)=AHOT4*N(4)
      A(11,5)=AHOT5*N(5)
      A(11,6)=AHOT6*N(6)
      A(11,7)=AHOT7*N(7)
      A(11,8)=AHOT8*N(8)
      A(11,9)=AHOT9*N(9)
```

```
      A(11,10)=-(HACALC)
      A(11,11)=(SCPNT*AT)/[illegible]
      A(11,12)=(HACALC)*ALOG(HTRUE/HCALC)
C     THIS FINISHES THE CALCULATIONS OF ELEMENTS OF REQUIRED MATRIX.
      DO 11 I=1,11
   11 B(I,1)=A(I,12)
      CALL MATINV(A,L,B,M,DET)
      DO 23 J=1,11
   23 X(J)=B(J,1)
C     HERE STARTS CHECKING OF X(I) AND INTERPOLATION IF ANY.
      CALL MAXMIN(X,L,XMAX,XMIN)
C     CALCULATIONS OF NEW VALUES TO BE USED,STARTS HERE.
      DO 890 I=1,9
      P(I)=X(I)+ALOG(N(I))
      N(I)=EXP(P(I))
  890 CONTINUE
      P(10)=X(10)+ALOG(ANEQW)
      ANEQW=EXP(P(10))
      P(11)=X(11)+ALOG(AT)
      CALCAT=EXP(P(11))
      PERRAT=((AT-CALCAT)/AT)*1[illegible]
      IF (PERRAT)1000,66,1001
 1000 IF (ABS(PERRAT).LE.1.25) GO TO 66
      IF (ABS(PERRAT).LE.55.00) GO TO 900
      GO TO 910
  900 INDEX=0
      DO 901 I=1,9
      N(I)=1.00
  901 CONTINUE
      ANEQW=1.00
      AT=AT+10.00
      GO TO 920
  910 INDEX=0
      DO 911 I=1,9
      N(I)=1.00
  911 CONTINUE
      ANEQW=1.00
      AT=AT+100.00
      GO TO 920
 1001 IF (ABS(PERRAT).LE.1.25) GO TO 66
      IF (ABS(PERRAT).LE.2[illegible]) GO TO 930
      GO TO 940
  930 INDEX=0
      DO 931 I=1,9
      N(I)=1.00
  931 CONTINUE
      ANEQW=1.00
      AT=AT-10.00
      GO TO 920
  940 INDEX=0
```

```
      DO 941 I=1,9
      N(I)=1.0D
  941 CONTINUE
      ANEQW=1.D
      AT=AT-100.D
  92  I=((AT-400.0)/20. )+
C     INTERPOLATION OF TABLE VALUES STARTS HERE.
      AAK1=AK1(I)+((AT-T(I))/(T(I+1)-T(I)))*(AK1(I+1)-AK1(I))
      AAK2=AK2(I)+((AT-T(I))/(T(I+1)-T(I)))*(AK2(I+1)-AK2(I))
      AAK3=AK3(I)+((AT-T(I))/(T(I+1)-T(I)))*(AK3(I+1)-AK3(I))
      AAK4=AK4(I)+((AT-T(I))/(T(I+1)-T(I)))*(AK4(I+1)-AK4(I))
      AAK5=AK5(I)+((AT-T(I))/(T(I+1)-T(I)))*(AK5(I+1)-AK5(I))
      AAK6=AK6(I)+((AT-T(I))/(T(I+1)-T(I)))*(AK6(I+1)-AK6(I))
      ACPT1=CPT1(I)+((AT-T(I))/(T(I+1)-T(I)))*(CPT1(I+1)-CPT1(I))
      ACPT2=CPT2(I)+((AT-T(I))/(T(I+1)-T(I)))*(CPT2(I+1)-CPT2(I))
      ACPT3= CPT3(I)+((AT T(I))/(T(I+1)-T(I)))*(CPT3(I+1)-CPT3(I))
      ACPT4=CPT4(I)+((AT-T(I))/(T(I+1)-T(I)))*(CPT4(I+1)-CPT4(I))
      ACPT5=CPT5(I)+((AT-T(I))/(T(I+1)-T(I)))*(CPT5(I+1)-CPT5(I))
      ACPT6=CPT6(I)+((AT-T(I))/(T(I+1)-T(I)))*(CPT6(I+1)-CPT6(I))
      ACPT7=CPT7(I)+((AT-T(I))/(T(I+1)-T(I)))*(CPT7(I+1)-CPT7(I))
      ACPT8=CPT8(I)+((AT-T(I))/(T(I+1)-T(I)))*(CPT8(I+1)-CPT8(I))
      ACPT9=CPT9(I)+((AT-T(I))/(T(I+1)-T(I)))*(CPT9(I+1)-CPT9(I))
      CPT1B=(ACPT1+CPT1(I))/2.
      CPT2B=(ACPT2+CPT2(I))/2.
      CPT3B=(ACPT3+CPT3(I))/2.
      CPT4B=(ACPT4+CPT4(I))/2.
      CPT5B=(ACPT5+CPT5(I))/2.
      CPT6B=(ACPT6+CPT6(I))/2.
      CPT7B=(ACPT7+CPT7(I))/2.
      CPT8B=(ACPT8+CPT8(I))/2.
      CPT9B=(ACPT9+CPT9(I))/2.
      AHOT1=HOT1(I)+((CPT1B*(AT-T(I)))/1000.00)
      AHOT2=HOT2(I)+((CPT2B*(AT-T(I)))/1000.00)
      AHOT3=HOT3(I)+((CPT3B*(AT-T(I)))/1000.00)
      AHOT4=HOT4(I)+((CPT4B*(AT-T(I)))/1000.00)
      AHOT5=HOT5(I)+((CPT5B*(AT-T(I)))/1000.00)
      AHOT6=HOT6(I)+((CPT6B*(AT-T(I)))/1000.00)
      AHOT7=HOT7(I)+((CPT7B*(AT-T(I)))/1000.00)
      AHOT8=HOT8(I)+((CPT8B*(AT-T(I)))/1000.00)
      AHOT9=HOT9(I)+((CPT9B*(AT-T(I)))/1000.00)
C     INTERPOLATION OF TABLE VALUES ENDS HERE.
      K=K+1
      IF (K.GT.3 ) GO TO 3
      GO TO 68
   66 CONTINUE
      NT=(N(1)+N(2)+N(3)+N(4)+N(5)+N(6)+N(7)+N(8)+N(9))/ANEQW
      DO 850 I=1,9
      NJ(I)=N(I)/ANEQW
      XJ(I)=NJ(I)/NT
  850 CONTINUE
```

```
      PRINT 851
  851 FORMAT (//1 X,* COMBUSTION CHAMBER PRESSURE     OFR *)
      PRINT 852,CPTRA,CFR(JJ)
  852 FORMAT (1X,2E14.5)
      PRINT 950
  950 FORMAT (* NO OF MOLES OF NITROGEN PRESENT AS IMPURITY *)
      PRINT 951,FACOIM
  951 FORMAT (1X,E12.4)
      PRINT 93
   93 FORMAT (* ABSOLUTE MAXIMUM ERROR   ABSOLUTE MINIMUM ERROR *)
      PRINT 29,XMAX,XMIN
   29 FORMAT (1X,2E11.4)
      PRINT 110
  110 FORMAT (* NEW VALUES OF N(I)S *)
      PRINT 120,(N(I),I=1,9)
  120 FORMAT (1X,9E12.4)
      PRINT 130
  130 FORMAT (* NEW VALUE OF EQ.FORMULA WEIGHT *)
      PRINT 140,ANEQW
  140 FORMAT (1X,E12.4)
      PRINT 150
  150 FORMAT (* NEW VALUE OF COMBUSTION TEMP. IN DEGREE KELVIN *)
      PRINT 160,CALCAT
  160 FORMAT (1X,E14.5)
      PRINT 800
  800 FORMAT (* INITIAL GUESS FOR COMBUSTION TEMPARATURE *)
      PRINT 820,AT
  820 FORMAT (1X,E14.5)
      PRINT 600
  600 FORMAT (* PER. ERROR IN COMBUSTION TEMPARATURE CALCULATION *)
      PRINT 610,PERRAT
  610 FORMAT (1X,E12.4)
      PRINT 860
  860 FORMAT (* MOLE FRACTIONS FOR VARIOUS SPECIES AT COMB TEMP *)
      PRINT 870,(XJ(I),I=1,9)
  870 FORMAT (1X,9E12.4)
      GO TO 3002
 3000 PRINT 3001
 3001 FORMAT (* NO OF ITERATIONS EXCEED THE SPECIFIED LIMIT *)
 3002 CONTINUE
 1111 CONTINUE
      STOP
      END
CIBFTC MAXMIN
      SUBROUTINE MAXMIN(X,L,XMAX,XMIN)
      DIMENSION X(11)
      XMAX=ABS(X(1))
      XMIN=XMAX
      DO 31 J=1,11
      IF (ABS(X(J)).GT.XMAX) XMAX=ABS(X(J))
```

```
      IF (ABS(X(J)).LT.XMIN) XMIN=ABS(X(J))
   31 CONTINUE
      RETURN
      END
CFETCH MATINV  CCS999
      SUBROUTINE  MATINV(C,N,B,M,DETERM)                               MAT0
      DIMENSION C(11,12),B(11,1),IPIVOT(11),INDEX(11,2)                MAT0
      DOUBLE PRECISION A(11,11),AMAX,T,SWAP,PIVOT                      MAT0
      DO 5 I=1,N                                                       MAT0
      DO 5 J=1,N                                                       MAT0
    5 A(I,J)=C(I,J)                                                    MAT0
   80 IF (AMAX-DABS(A(J,K)))85,100,100                                 MAT0
   95 AMAX=DABS(A(J,K))                                                MAT0
  320 DETERM=DETERM*PIVOT/DABS(PIVOT)                                  MAT0
      DO 780 I=1,N                                                     MAT0
      DO 780 J=1,N                                                     MAT0
  780 C(I,J)=A(I,J)                                                    MAT0
CDONE
CENTRY
C     NOW FOLLOWS THE NECESSARY DATA CARDS.
```

Calculated datas are tabulates as under.

TABLE 33

Theoretically Calculated Data For Equilibrium Temperature and Composition Of Product Gas For Hydrogen, Oxygen, Nitrogen Propellant System Chemical Rocket Motor, For

$p_c$ = 3.40 Ata  O/F Ratio = 2.00

$T_c$ Calculated = 3390.0 °K  A Calculated = 0.3742

| SPECIES | NUMBER OF MOLES $n_i$ | MOLE FRACTION $X_j$ |
|---|---|---|
| $H_2O$ | 0.6393 | 0.1880 |
| OH | 0.08411 | 0.02474 |
| NO | 0.0004784 | 0.0001407 |
| $H_2$ | 1.9730 | 0.5804 |
| $O_2$ | 0.002277 | 0.0006698 |
| $N_2$ | 0.002900 | 0.0008528 |
| H | 0.6773 | 0.1992 |
| O | 0.01987 | 0.005844 |
| N | 0.0004574 | 0.0001345 |

TABLE 34

Theoretically Calculated Data For Equilibrium Temperature and Composition Of Product Gas For Hydrogen, Oxygen, Nitrogen Propellant System Chemical Rocket Motor, For

$p_c$ = 3.40 Ata O/F Ratio = 4.00

$T_c$ Calculated = 3607.4 °K $\Lambda$ Calculated = 0.6758

| SPECIES | NUMBER OF MOLES $n_i$ | MOLE FRACTION $X_j$ |
|---|---|---|
| $H_2O$ | 0.[illegible]085 | 0.2672 |
| OH | 0.2648 | 0.07789 |
| NO | 0.001755 | 0.0005161 |
| $H_2$ | 1.2488 | 0.3672 |
| $O_2$ | 0.03011 | 0.008855 |
| $N_2$ | 0.002113 | 0.0006215 |
| H | 0.8272 | 0.2433 |
| O | 0.1162 | 0.03417 |
| N | 0.0007764 | 0.0002284 |

TABLE 35

Theoretically Calculated Data For Equilibrium Temperature and Composition Of Product Gas For Hydrogen, Oxygen, Nitrogen Propellant System Chemical Rocket Motor, For

$p_c$ = 3.40 Ata O/F Ratio = 6.00

$T_c$ Calculated = 3629.85 °K $\Lambda$ Calculated = 0.9210

| SPECIES | NUMBER OF MOLES $N_i$ | MOLE FRACTION $X_j$ |
|---|---|---|
| $H_2O$ | 0.9886 | 0.2907 |
| NO | 0.002826 | 0.0008310 |
| $H_2$ | 0.8660 | 0.2547 |
| $O_2$ | 0.09643 | 0.02836 |
| $N_2$ | 0.001550 | 0.0004559 |
| H | 0.7865 | 0.2313 |
| O | 0.2409 | 0.07085 |
| N | 0.0008284 | 0.0002436 |
| OH | 0.4168 | 0.1226 |

TABLE 36

Theoretically Calculated Data For Equilibrium Temperature and Composition Of Product Gas For Hydrogen, Oxygen, Nitrogen Propellant System Chemical Rocket Motor, For

$P_c$ = 3.40 Ata　　O/F Ratio = 8.00

$T_c$ Calculated = 3674.5 °K　　A Calculated = 1.1350

| SPECIES | NUMBER OF MOLES $n_i$ | MOLE FRACTION $X_j$ |
|---|---|---|
| $H_2O$ | 1.0350 | 0.3045 |
| OH | 0.5068 | 0.1491 |
| NO | 0.003576 | 0.001052 |
| $H_2$ | 0.6426 | 0.1890 |
| $O_2$ | 0.1921 | 0.05651 |
| $N_2$ | 0.001246 | 0.0003665 |
| H | 0.6775 | 0.1993 |
| O | 0.3401 | 0.1000 |
| N | 0.0007427 | 0.0002184 |

TABLE 37

Theoretically Calculated Data For Equilibrium Temperature and Composition Of Product Gas For Hydrogen, Oxygen, Nitrogen Propellant System Chemical Rocket Motor, For

$P_c$ = 6.80 Ata　　O/F Ratio = 2.00

$T_c$ Calculated = 3532.5 °K　　A Calculated = 0.7484

| SPECIES | NUMBER OF MOLES $n_i$ | MOLE FRACTION $X_j$ |
|---|---|---|
| $H_2O$ | 1.2640 | 0.1860 |
| OH | 0.1833 | 0.02696 |
| NO | 0.001040 | 0.0001529 |
| $H_2$ | 3.9620 | 0.5827 |
| $O_2$ | 0.004282 | 0.0006297 |
| $N_2$ | 0.005664 | 0.0008329 |
| H | 1.3380 | 0.1968 |
| O | 0.03946 | 0.005802 |
| N | 0.001105 | 0.0001625 |

TABLE 38

Theoretically Calculated Data For Equilibrium Temperature and Composition Of Product Gas For Hydrogen, Oxygen, Nitrogen Propellant System Chemical Rocket Motor, For

$p_c$ = 6.80 Ata | O/F Ratio = 4.00

$T_c$ Calculated = 3742.8 °K | A Calculated = 1.3500

| SPECIES | NUMBER OF MOLES $n_i$ | MOLE FRACTION $X_j$ |
|---|---|---|
| $H_2O$ | 1.7750 | 0.2611 |
| OH | 0.5711 | 0.08389 |
| NO | 0.003795 | 0.0005581 |
| $H_2$ | 2.5220 | 0.3708 |
| $O_2$ | 0.05837 | 0.008584 |
| $N_2$ | 0.003955 | 0.0005817 |
| H | 1.6320 | 0.2400 |
| O | 0.2324 | 0.03417 |
| N | 0.001790 | 0.0002632 |

TABLE 39

Theoretically Calculated Data For Equilibrium Temperature and Composition Of Product Gas For Hydrogen, Oxygen, Nitrogen Propellant System Chemical Rocket Motor, For

$p_c$ = 6.80 Ata | O/F Ratio = 6.00

$T_c$ Calculated = 3796.8 °K | A Calculated = 1.846

| SPECIES | NUMBER OF MOLES $n_i$ | MOLE FRACTION $X_j$ |
|---|---|---|
| $H_2O$ | 1.9560 | 0.2876 |
| OH | 0.8623 | 0.1268 |
| NO | 0.006132 | 0.0009017 |
| $H_2$ | 1.7760 | 0.2612 |
| $O_2$ | 0.1957 | 0.02877 |
| $N_2$ | 0.002820 | 0.0004146 |
| H | 1.5230 | 0.2239 |
| O | 0.4776 | 0.07022 |
| N | 0.001770 | 0.0002602 |

TABLE 40

Theoretically Calculated Data for Equilibrium Temperature and Composition Of Product Gas For Hydrogen, Oxygen, Nitrogen Propellant System Chemical Rocket Motor, For

$p_c$ = 6.80 Ata    O/F Ratio = 8.00

$T_c$ Calculated = 3802.4 °K    A Calculated = 2.260

| SPECIES | NUMBER OF MOLES $n_i$ | MOLE FRACTION $X_j$ |
|---|---|---|
| $H_2O$ | 2.0030 | 0.2946 |
| OH | 1.0550 | 0.15520 |
| NO | 0.007608 | 0.001119 |
| $H_2$ | 1.3200 | 0.1942 |
| $O_2$ | 0.3849 | 0.05661 |
| $N_2$ | 0.002174 | 0.0003197 |
| H | 1.3400 | 0.1971 |
| O | 0.6853 | 0.1008 |
| N | 0.001608 | 0.0002364 |

TABLE 41

Theoretically Calculated Data For Equilibrium Temperature and Composition of Product Gas For Hydrogen, Oxygen, Nitrogen Propellant System Chemical Rocket Motor, For

$p_c$ = 10.20 Ata    O/F Ratio = 2.00

$T_c$ Calculated = 3606.4 °K    A Calculated = 1.120

| SPECIES | NUMBER OF MOLES $n_i$ | MOLE FRACTION $X_j$ |
|---|---|---|
| $H_2O$ | 1.8700 | 0.1833 |
| OH | 0.2919 | 0.02862 |
| NO | 0.001750 | 0.0001716 |
| $H_2$ | 5.9400 | 0.5824 |
| $O_2$ | 0.007035 | 0.0006897 |
| $N_2$ | 0.008235 | 0.0008123 |
| H | 2.0150 | 0.1976 |
| O | 0.06352 | 0.006227 |
| N | 0.001848 | 0.0001811 |

TABLE 42

Theoretically Calculated Data For Equilibrium Temperature and Composition Of Product Gas For Hydrogen, Oxygen Nitrogen Propellant System Chemical Rocket Motor, For

$p_c$ = 10.20 Ata  O/F Ratio = 4.00

$T_c$ Calculated = 3835.7 °K  A Calculated = 2.0240

| SPECIES | NUMBER OF MOLES $n_i$ | MOLE FRACTION $X_j$ |
|---|---|---|
| $H_2O$ | 2.6280 | 0.2577 |
| OH | 0.8852 | 0.08678 |
| NO | 0.006026 | 0.0005908 |
| $H_2$ | 3.8160 | 0.3741 |
| $O_2$ | 0.08855 | 0.008681 |
| $N_2$ | 0.005676 | 0.0005564 |
| H | 2.4170 | 0.2370 |
| O | 0.3509 | 0.03440 |
| N | 0.002860 | 0.0002804 |

TABLE 43

Theoretically Calculated Data For Equilibrium Temperature and Composition Of Product Gas For Hydrogen, Oxygen, Nitrogen Propellant System Chemical Rocket Motor, For

$p_c$ = 10.20 Ata  O/F Ratio = 6.00

$T_c$ Calculated = 3902.4 °K  A Calculated = 2.7720

| SPECIES | NUMBER OF MOLES $n_i$ | MOLE FRACTION $X_j$ |
|---|---|---|
| $H_2O$ | 2.9110 | 0.2853 |
| OH | 1.3580 | 0.1332 |
| NO | 0.009429 | 0.0009243 |
| $H_2$ | 2.6750 | 0.2622 |
| $O_2$ | 0.2808 | 0.02752 |
| $N_2$ | 0.004025 | 0.0003946 |
| H | 2.2560 | 0.2212 |
| O | 0.7040 | 0.06901 |
| N | 0.002850 | 0.0002794 |

TABLE 44

Theoretically Calculated Data For Equilibrium Temperature and Composition Of Product Gas For Hydrogen, Oxygen, Nitrogen Propellant System Chemical Rocket Motor, For

$p_c$ = 10.20 Ata    O/F Ratio 8.00

$T_c$ Calculated = 3909.4 °K    A Calculated = 3.3960

| SPECIES | NUMBER OF MOLES $n_i$ | MOLE FRACTION $X_j$ |
|---|---|---|
| $H_2O$ | 2.9890 | 0.2930 |
| OH | 1.6560 | 0.1624 |
| NO | 0.01170 | 0.001147 |
| $H_2$ | 1.9860 | 0.1947 |
| $O_2$ | 0.5606 | 0.05496 |
| $N_2$ | 0.003062 | 0.0003002 |
| H | 1.9780 | 0.1939 |
| O | 1.0130 | 0.09936 |
| N | 0.002550 | 0.0002500 |

TABLE 45

Theoretically Calculated Data For Equilibrium Temperature and Composition Of Product Gas For Hydrogen, Oxygen, Nitrogen Propellant System Chemical Rocket Motor, For

$p_c$ = 13.60 Ata    O/F Ratio = 2.00

$T_c$ Calculated = 3686.8 °K    A Calculated = 1.4970

| SPECIES | NUMBER OF MOLES $n_i$ | MOLE FRACTION $X_j$ |
|---|---|---|
| $H_2O$ | 2.4910 | 0.1832 |
| OH | 0.3988 | 0.02932 |
| NO | 0.002393 | 0.0001759 |
| $H_2$ | 7.9610 | 0.5853 |
| $O_2$ | 0.009021 | 0.0006633 |
| $N_2$ | 0.01098 | 0.0008071 |
| H | 2.6420 | 0.1943 |
| O | 0.08252 | 0.006068 |
| N | 0.002593 | 0.0001906 |

TABLE 46

Theoretically Calculated Data For Equilibrium Temperature and Composition Of Product Gas For Hydrogen, Oxygen, Nitrogen Propellant System Chemical Rocket Motor, For

$p_c$ = 13.60 Ata  O/F Ratio = 4.00

$T_c$ Calculated = 3927.4 °K  A Calculated = 2.7030

| SPECIES | NUMBER OF MOLES $n_i$ | MOLE FRACTION $X_j$ |
|---|---|---|
| $H_2O$ | 3.5000 | 0.2573 |
| OH | 1.2070 | 0.8878 |
| NO | 0.008259 | 0.0006073 |
| $H_2$ | 5.1220 | 0.3766 |
| $O_2$ | 0.11550 | 0.008495 |
| $N_2$ | 0.007404 | 0.0005444 |
| H | 3.1760 | 0.2335 |
| O | 0.4601 | 0.03383 |
| N | 0.003965 | 0.0002916 |

TABLE 47

Theoretically Calculated Data For Equilibrium Temperature and Composition Of Product Gas For Hydrogen, Oxygen, Nitrogen Propellant System Chemical Rocket Motor, For

$p_c$ = 13.60 Ata  O/F Ratio = 6.00

$T_c$ Calcuated = 3964.2 °K  A Calculated = 3.6910

| SPECIES | NUMBER OF MOLES $n_i$ | MOLE FRACTION $X_j$ |
|---|---|---|
| $H_2O$ | 3.8280 | 0.2814 |
| OH | 1.8200 | 0.1338 |
| NO | 0.01308 | 0.0009618 |
| $H_2$ | 3.6130 | 0.2656 |
| $O_2$ | 0.3858 | 0.02836 |
| $N_2$ | 0.005064 | 0.0003723 |
| H | 2.9830 | 0.2193 |
| O | 0.9490 | 0.06977 |
| N | 0.003857 | 0.0002836 |

TABLE 48

Theoretically Calculated Data For Equilibrium Temperature and Composition Of Product Gas For Hydrogen, Oxygen, Nitrogen Propellant System Chemical Rocket Motor, For

$p_c$ = 13.60 Ata C/F Ratio = 8.00

$T_c$ Calculated = 4017.5 °K A Calculated = 4.5450

| SPECIES | NUMBER OF MOLES $n_i$ | MOLE FRACTION $X_j$ |
|---|---|---|
| $H_2O$ | 4.0070 | 0.2947 |
| OH | 2.2080 | 0.1623 |
| NO | 0.01611 | 0.001185 |
| $H_2$ | 2.6910 | 0.1979 |
| $O_2$ | 0.7622 | 0.05604 |
| $N_2$ | 0.00388 | 0.0002859 |
| H | 2.5750 | 0.1893 |
| O | 1.3340 | 0.09808 |
| N | 0.003380 | 0.0002485 |

B.2 Frozen Flow Approach

Solution to the problem discussed in Part 2.3, Section II is attempted here. Calculated readings mentioned in Tables (33) to (48) are to be taken as input data for these calculations. Necessary thermodynamic tables for species are to be taken from Reference (19). A suitable temperature interval of 200 °K is chosen. Attached herewith is the developed computer programme for system IBM 7044/1401 which will provide the method of solution. Error limits were modified at times to reach convergence. All calculations are carried out for small $\alpha$ and $p_e$ = Atmospheric pressure.

```
CUTE       WATFOR,REWIND
TC MAIN
  PERFORMANCE ANALYSIS OF GAS ROCKET MOTOR USING H2-O2
  PROPELLANTS WITH FROZEN FLOW TECHNIQUE.
  READING OF VARIOUS VALUES MENTIONED IN DIMENSION STATEMENT ARE
  FROM STANDARD TABLES AVAILABLES.
   UNITS OF ENTHALPY ARE KCAL/MOLE
  UNITS OF SP HEATS ARE CAL/OK MOLE.
  UNITS OF ENTROPY ARE CAL/OK-MOLE
  DIMENSION HOT1(24),HOT2(24),HOT3(24),HOT4(24),HOT5(24),HOT6(24),
 1HOT7(24),HOT8(24),HOT9(24),CPT1(24),CPT2(24),CPT3(24),CPT4(24),
 1CPT5(24),CPT6(24),CPT7(24),CPT8(24),CPT9(24),SOTC1(24),SOTC2(24)
  DIMENSION SOTC3(24),SOTC4(24),SOTC5(24),SOTC6(24),SOTC7(24),
 1SOTC8(24),SOTC9(24),HOTE1(24),HOTE2(24),HOTE3(24),HOTE4(24),
 1HOTE5(24),HOTE6(24),HOTE7(24),HOTE8(24),HOTE9(24),CPTE1(24),
 1CPTE2(24),CPTE3(24),CPTE4(24),CPTE5(24),CPTE6(24),CPTE7(24)
  DIMENSION CPTE8(24),CPTE9(24),SOTE1(24),SOTE2(24),SOTE3(24),
 1SOTE4(24),SOTE5(24),SOTE6(24),SOTE7(24),SOTE8(24),SOTE9(24),T(24),
 1ET(24),TC(4),XJ(4,9),OFR(4),W(9)
  READ 4,(T(I),I=1,24)
4 FORMAT  (8F10.1)
  READ 12,(HOT1(I),I=1,24)
2 FORMAT (8F10.4)
  READ 12,(HOT2(I),I=1,24)
  READ 12,(HOT3(I),I=1,24)
  READ 12,(HOT4(I),I=1,24)
  READ 12,(HOT5(I),I=1,24)
  READ 12,(HOT6(I),I=1,24)
  READ 12,(HOT7(I),I=1,24)
  READ 12,(HOT8(I),I=1,24)
  READ 12,(HOT9(I),I=1,24)
  READ 14,(CPT1(I),I=1,24)
4 FORMAT (8F10.3)
  READ 14,(CPT2(I),I=1,24)
  READ 14,(CPT3(I),I=1,24)
  READ 16,(CPT4(I),I=1,24)
5 FORMAT (8F10.4)
  READ 14,(CPT5(I),I=1,24)
  READ 16,(CPT6(I),I=1,24)
  READ 14,(CPT7(I),I=1,24)
  READ 16,(CPT8(I),I=1,24)
  READ 16,(CPT9(I),I=1,24)
  READ 18,(OFR(JJ),JJ=1,4)
3 FORMAT (4F10.1)
  READ 20,(SOTC1(I),I=1,24)
0 FORMAT (8F10.4)
  READ 20,(SOTC2(I),I=1,24)
  READ 20,(SOTC3(I),I=1,24)
  READ 20,(SOTC4(I),I=1,24)
  READ 20,(SOTC5(I),I=1,24)
```

```
      READ 20,(SOTC6(I),I=1,24)
      READ 20,(SOTC7(I),I=1,24)
      READ 20,(SOTC8(I),I=1,24)
      READ 20,(SOTC9(I),I=1,24)
      READ 4,(ET(I),I=1,24)
      READ 12,(HOTE1(I),I=1,24)
      READ 12,(HOTE2(I),I=1,24)
      READ 12,(HOTE3(I),I=1,24)
      READ 12,(HOTE4(I),I=1,24)
      READ 12,(HOTE5(I),I=1,24)
      READ 12,(HOTE6(I),I=1,24)
      READ 12,(HOTE7(I),I=1,24)
      READ 12,(HOTE8(I),I=1,24)
      READ 12,(HOTE9(I),I=1,24)
      READ  14,(CPTE1(I),I=1,24)
      READ  14,(CPTE2(I),I=1,24)
      READ  14,(CPTE3(I),I=1,24)
      READ 16,(CPTE4(I),I=1,24)
      READ 14,(CPTE5(I),I=1,24)
      READ 16,(CPTE6(I),I=1,24)
      READ 14,(CPTE7(I),I=1,24)
      READ 16,(CPTE8(I),I=1,24)
      READ 16,(CPTE9(I),I=1,24)
      READ 20,(SOTE1(I),I=1,24)
      READ 20,(SOTE2(I),I=1,24)
      READ 20,(SOTE3(I),I=1,24)
      READ 20,(SOTE4(I),I=1,24)
      READ 20,(SOTE5(I),I=1,24)
      READ 20,(SOTE6(I),I=1,24)
      READ 20,(SOTE7(I),I=1,24)
      READ 20,(SOTE8(I),I=1,24)
      READ 20,(SOTE9(I),I=1,24)
      READ 22,(TC(I),I=1,4)
   22 FORMAT (4F10.4)
      READ 24,((XJ(I,II),II=1,9),I=1,4)
   24 FORMAT (3(E10.4,2X))
      READ 26,(W(I),I=1,9)
   26 FORMAT (9F6.1)
C     THIS FINISHES THE TABLE READING
C     CALCULATION OF EXIT TEMPRATURE STARTS
C     ATC=AT
      UGC=1.9857
      CPTRA=3.40
      RHS=UGC*ALOG(CPTRA)
      DO 1111 JJ=1,4
      ATE=3000.00
  119 ATC=TC(JJ)
      I=((ATC-400.00)/200.00)+1.00
      J=((ATE-400.00)/200.00)+1.00
      ASOTC1=SOTC1(I)+((ATC-T(I))/(T(I+1)-T(I)))*(SOTC1(I+1)-SOTC1(I))
```

```
      ASOTC2=SOTC2(I)+((ATC-T(I))/(T(I+1)-T(I)))*(SOTC2(I+1)-SOTC2(I))
      ASOTC3=SOTC3(I)+((ATC-T(I))/(T(I+1)-T(I)))*(SOTC3(I+1)-SOTC3(I))
      ASOTC4=SOTC4(I)+((ATC-T(I))/(T(I+1)-T(I)))*(SOTC4(I+1)-SOTC4(I))
      ASOTC5=SOTC5(I)+((ATC-T(I))/(T(I+1)-T(I)))*(SOTC5(I+1)-SOTC5(I))
      ASOTC6=SOTC6(I)+((ATC-T(I))/(T(I+1)-T(I)))*(SOTC6(I+1)-SOTC6(I))
      ASOTC7=SOTC7(I)+((ATC-T(I))/(T(I+1)-T(I)))*(SOTC7(I+1)-SOTC7(I))
      ASOTC8=SOTC8(I)+((ATC-T(I))/(T(I+1)-T(I)))*(SOTC8(I+1)-SOTC8(I))
      ASOTC9=SOTC9(I)+((ATC-T(I))/(T(I+1)-T(I)))*(SOTC9(I+1)-SOTC9(I))
      ASOTE1=SOTE1(J)+((ATE-ET(J))/(ET(J+1)-ET(J)))*(SOTE1(J+1)-SOTE2(J)
     1)
      ASOTE2=SOTE2(J)+((ATE-ET(J))/(ET(J+1)-ET(J)))*(SOTE2(J+1)-SOTE2(J)
     1)
      ASOTE3=SOTE3(J)+((ATE-ET(J))/(ET(J+1)-ET(J)))*(SOTE3(J+1)-SOTE3(J)
     1)
      ASOTE4=SOTE4(J)+((ATE-ET(J))/(ET(J+1)-ET(J)))*(SOTE4(J+1)-SOTE4(J)
     1)
      ASOTE5=SOTE5(J)+((ATE-ET(J))/(ET(J+1)-ET(J)))*(SOTE5(J+1)-SOTE5(J)
     1)
      ASOTE6=SOTE6(J)+((ATE-ET(J))/(ET(J+1)-ET(J)))*(SOTE6(J+1)-SOTE6(J)
     1)
      ASOTE7=SOTE7(J)+((ATE-ET(J))/(ET(J+1)-ET(J)))*(SOTE7(J+1)-SOTE7(J)
     1)
      ASOTE8=SOTE8(J)+((ATE-ET(J))/(ET(J+1)-ET(J)))*(SOTE8(J+1)-SOTE8(J)
     1)
      ASOTE9=SOTE9(J)+((ATE-ET(J))/(ET(J+1)-ET(J)))*(SOTE9(J+1)-SOTE9(J)
     1)
      TERM1=(XJ(JJ,1))*(ASOTC1-ASOTE1)
      TERM2=(XJ(JJ,2))*(ASOTC2-ASOTE2)
      TERM3=(XJ(JJ,3))*(ASOTC3-ASOTE3)
      TERM4=(XJ(JJ,4))*(ASOTC4-ASOTE4)
      TERM5=(XJ(JJ,5))*(ASOTC5-ASOTE5)
      TERM6=(XJ(JJ,6))*(ASOTC6-ASOTE6)
      TERM7=(XJ(JJ,7))*(ASOTC7-ASOTE7)
      TERM8=(XJ(JJ,8))*(ASOTC8-ASOTE8)
      TERM9=(XJ(JJ,9))*(ASOTC9-ASOTE9)
      SUMTS=TERM1+TERM2+TERM3+TERM4+TERM5+TERM6+TERM7+TERM8+TERM9
      PERROR=((SUMTS-RHS)/RHS)*100.00
      PRINT 600
  600 FORMAT (* PERCENTAGE ERROR *)
      PRINT 601,PERROR
  601 FORMAT (1X,E14.5)
      IF (ABS(PERROR).GE.25.00) GO TO 120
      IF (ABS(PERROR).LE.2.00) GO TO 122
      ATE=ATE-10.00
      GO TO 119
  120 ATE=ATE-100.00
      GO TO 119
  122 PRINT 132
  132 FORMAT (* RHS   SUMTS *)
      PRINT 134,RHS,SUMTS
```

```
  134 FORMAT (1X,2E14.5)
C      CALCULATIONS FOR EXIT TEMPRATURE END
C      CALCULATION OF EXHAUST VELOCITY STARS
       AVMWT=XJ(JJ,1)*W(1)+XJ(JJ,2)*W(2)+XJ(JJ,3)*W(3)+XJ(JJ,4)*W(4)+
      1XJ(JJ,5)*W(5)+XJ(JJ,6)*W(6)+XJ(JJ,7)*W(7)+XJ(JJ,8)*W(8)+XJ(JJ,9)*
      1W(9)
       ACPT1=CPT1(I)+((ATC-T(I))/(T(I+1)-T(I)))*(CPT1(I+1)-CPT1(I))
       ACPT2=CPT2(I)+((ATC-T(I))/(T(I+1)-T(I)))*(CPT2(I+1)-CPT2(I))
       ACPT3=CPT3(I)+((ATC-T(I))/(T(I+1)-T(I)))*(CPT3(I+1)-CPT3(I))
       ACPT4=CPT4(I)+((ATC-T(I))/(T(I+1)-T(I)))*(CPT4(I+1)-CPT4(I))
       ACPT5=CPT5(I)+((ATC-T(I))/(T(I+1)-T(I)))*(CPT5(I+1)-CPT5(I))
       ACPT6=CPT6(I)+((ATC-T(I))/(T(I+1)-T(I)))*(CPT6(I+1)-CPT6(I))
       ACPT7=CPT7(I)+((ATC-T(I))/(T(I+1)-T(I)))*(CPT7(I+1)-CPT7(I))
       ACPT8=CPT8(I)+((ATC-T(I))/(T(I+1)-T(I)))*(CPT8(I+1)-CPT8(I))
       ACPT9=CPT9(I)+((ATC-T(I))/(T(I+1)-T(I)))*(CPT9(I+1)-CPT9(I))
       CPT1B=(ACPT1+CPT1(I))/2.0
       CPT2B=(ACPT2+CPT2(I))/2.0
       CPT3B=(ACPT3+CPT3(I))/2.0
       CPT4B=(ACPT4+CPT4(I))/2.0
       CPT5B=(ACPT5+CPT5(I))/2.0
       CPT6B=(ACPT6+CPT6(I))/2.0
       CPT7B=(ACPT7+CPT7(I))/2.0
       CPT8B=(ACPT8+CPT8(I))/2.0
       CPT9B=(ACPT9+CPT9(I))/2.0
       AHOT1=HOT1(I)+((CPT1B*(ATC-T(I)))/1000.00)
       AHOT2=HOT2(I)+((CPT2B*(ATC-T(I)))/1000.00)
       AHOT3=HOT3(I)+((CPT3B*(ATC-T(I)))/1000.00)
       AHOT4=HOT4(I)+((CPT4B*(ATC-T(I)))/1000.00)
       AHOT5=HOT5(I)+((CPT5B*(ATC-T(I)))/1000.00)
       AHOT6=HOT6(I)+((CPT6B*(ATC-T(I)))/1000.00)
       AHOT7=HOT7(I)+((CPT7B*(ATC-T(I)))/1000.00)
       AHOT8=HOT8(I)+((CPT8B*(ATC-T(I)))/1000.00)
       AHOT9=HOT9(I)+((CPT9B*(ATC-T(I)))/1000.00)
       ACPTE1=CPTE1(J)+((ATE-ET(J))/(ET(J+1)-ET(J)))*(CPTE1(J+1)-CPTE1(J)
      1)
       ACPTE2=CPTE2(J)+((ATE-ET(J))/(ET(J+1)-ET(J)))*(CPTE2(J+1)-CPTE2(J)
      1)
       ACPTE3=CPTE3(J)+((ATE-ET(J))/(ET(J+1)-ET(J)))*(CPTE3(J+1)-CPTE3(J)
      1)
       ACPTE4=CPTE4(J)+((ATE-ET(J))/(ET(J+1)-ET(J)))*(CPTE4(J+1)-CPTE4(J)
      1)
       ACPTE5=CPTE5(J)+((ATE-ET(J))/(ET(J+1)-ET(J)))*(CPTE5(J+1)-CPTE5(J)
      1)
       ACPTE6=CPTE6(J)+((ATE-ET(J))/(ET(J+1)-ET(J)))*(CPTE6(J+1)-CPTE6(J))
      1)
       ACPTE7=CPTE7(J)+((ATE-ET(J))/(ET(J+1)-ET(J)))*(CPTE7(J+1)-CPTE7(J)
      1)
       ACPTE8=CPTE8(J)+((ATE-ET(J))/(ET(J+1)-ET(J)))*(CPTE8(J+1)-CPTE8(J)
      1)
       ACPTE9=CPTE9(J)+((ATE-ET(J))/(ET(J+1)-ET(J)))*(CPTE9(J+1)-CPTE9(J)
```

```
     1)
      CPTE1B=(ACPTE1+CPTE1(J))/2.0
      CPTE2B=(ACPTE2+CPTE2(J))/2.0
      CPTE3B=(ACPTE3+CPTE3(J))/2.0
      CPTE4B=(ACPTE4+CPTE4(J))/2.0
      CPTE5B=(ACPTE5+CPTE5(J))/2.0
      CPTE6B=(ACPTE6+CPTE6(J))/2.0
      CPTE7B=(ACPTE7+CPTE7(J))/2.0
      CPTE8B=(ACPTE8+CPTE8(J))/2.0
      CPTE9B=(ACPTE9+CPTE9(J))/2.0
      AHOTE1=HOTE1(J)+((CPTE1B*(ATE-ET(J)))/1000.00)
      AHOTE2=HOTE2(J)+((CPTE2B*(ATE-ET(J)))/1000.00)
      AHOTE3=HOTE3(J)+((CPTE3B*(ATE-ET(J)))/1000.00)
      AHOTE4=HOTE4(J)+((CPTE4B*(ATE-ET(J)))/1000.00)
      AHOTE5=HOTE5(J)+((CPTE5B*(ATE-ET(J)))/1000.00)
      AHOTE6=HOTE6(J)+((CPTE6B*(ATE-ET(J)))/1000.00)
      AHOTE7=HOTE7(J)+((CPTE7B*(ATE-ET(J)))/1000.00)
      AHOTE8=HOTE8(J)+((CPTE8B*(ATE-ET(J)))/1000.00)
      AHOTE9=HOTE9(J)+((CPTE9B*(ATE-ET(J)))/1000.00)
      TERMH1=(XJ(JJ,1))*(AHOT1-AHOTE1)
      TERMH2=(XJ(JJ,2))*(AHOT2-AHOTE2)
      TERMH3=(XJ(JJ,3))*(AHOT3-AHOTE3)
      TERMH4=(XJ(JJ,4))*(AHOT4-AHOTE4)
      TERMH5=(XJ(JJ,5))*(AHOT5-AHOTE5)
      TERMH6=(XJ(JJ,6))*(AHOT6-AHOTE6)
      TERMH7=(XJ(JJ,7))*(AHOT7-AHOTE7)
      TERMH8=(XJ(JJ,8))*(AHOT8-AHOTE8)
      TERMH9=(XJ(JJ,9))*(AHOT9-AHOTE9)
      SUMTH=TERMH1+TERMH2+TERMH3+TERMH4+TERMH5+TERMH6+TERMH7+TERMH8+
     1TERMH9
C     PERFORMANCE PARAMETER CALCULATIONS FOR SMALL ALPHA AND PEXIT=PATM
C     MKS UNITS ARE USED HERE.
      VEXIT=((SQRT(SUMTH))*(SQRT(8370.0)))/(SQRT(AVMWT))*SQRT(1000.0)
      EVEXIT=VEXIT
      SPIMP=VEXIT/9.81
      GSTAR=(ALOG(CPTRA))/(ALOG(CPTRA)-ALOG((ATC/ATE)))
      CGSTAR=(2.0/(GSTAR+1.))**((GSTAR+1.)/(2.0*(GSTAR-1.0)))
      CHVELO=(1.0/CGSTAR)*SQRT((UGC*ATC)/(AVMWT*GSTAR))*SQRT(4185.0)
      THCOFF=EVEXIT/CHVELO
      PRINT 127
  127 FORMAT (//10X,* COMBUSTION CHAMBER PRESSURE   OFR *)
      PRINT 131,CPTRA,CFR(JJ)
  131 FORMAT (1X,2E14.5)
      PRINT 124
  124 FORMAT (* COMBUSTION TEMPARATURE   EXIT TEMPARATURE *)
      PRINT 126,TC(JJ),ATE
  126 FORMAT (1X,2E14.5)
      PRINT 136
  136 FORMAT (* AVARAGE MOLECULAR WEIGHT IN GMS/MOLE *)
      PRINT 138,AVMWT
```

```
  138 FORMAT (1X,E14.5)
      PRINT 128
  128 FORMAT (* VEXIT EVEXIT SPIMP GSTAR CHVELO THCOFF *)
      PRINT 130,VEXIT,EVEXIT,SPIMP,GSTAR,CHVELO,THCOFF
  130 FORMAT (1X,6E14.6)
 1111 CONTINUE
      STOP
      END
CENTRY
C     NOW FOLLOWS THE NECESSARY DATA CARDS.
```

Necessary data for calculations is taken from the Tables (33), (34), (35) and (36). Calculated data is tabulated as under.

TABLE 49

Theoretically Calculated Data For Performance Parameters For Hydrogen, Oxygen, Nitrogen Propellant System Chemical Rocket Motor At Different O/F Ratios With Frozen Flow Approach, For $p_c$ = 3.40 Ata.

| O/F RATIO | | 2 | 4 | 6 | 8 |
|---|---|---|---|---|---|
| $T_c$ | °K | 3393.00 | 3607.40 | 3629.90 | 3674.50 |
| $T_e$ | °K | 2470.00 | 2640.00 | 2640.00 | 2650.00 |
| $\overline{W}^*$ | Gms/Mole | 5.3095 | 7.9776 | 10.1400 | 12.0460 |
| $v_e$ | M/Sec. | 3575.29 | 3006.66 | 2699.93 | 2533.54 |
| $u_{eq}$ | M/Sec. | 3575.29 | 3006.66 | 2699.93 | 2533.54 |
| $I_{sp}$ | Kgf Sec./ Kgm | 364.453 | 306.490 | 275.222 | 258.261 |
| $\overline{\gamma}^*$ | | 1.35033 | 1.34250 | 1.35170 | 1.36442 |
| C* | M/Sec. | 3407.95 | 2872.57 | 2549.80 | 2346.02 |
| $C_F$ | | 1.04910 | 1.04668 | 1.05888 | 1.07993 |

Necessary data for calculations is taken from the Tables (37), (38), (39) and (40). Calculated data is tabulated as under.

TABLE 50

Theoretically Calculated Data For Performance Parameters For Hydrogen, Oxygen, Nitrogen Propellant System Chemical Rocket Motor At Different O/F Ratios With Frozen Flow Approach, For $p_c$ = 6.80 Ata.

| O/F RATIO | | 2 | 4 | 6 | 8 |
|---|---|---|---|---|---|
| $T_c$ | °K | 3532.50 | 3742.80 | 3796.80 | 3802.40 |
| $T_e$ | °K | 2240.00 | 2280.00 | 2290.00 | 2290.00 |
| $\overline{W}^*$ | Gms/Mole | 5.3037 | 7.9672 | 10.1650 | 11.9970 |
| $v_e$ | M/Sec. | 4226.13 | 3686.07 | 3322.35 | 3069.96 |
| $u_{eq}$ | M/Sec. | 4226.13 | 3686.07 | 3322.35 | 3069.96 |
| $I_{sp}$ | Kgf Sec./ Kgm | 430.798 | 375.746 | 338.670 | 312.942 |
| $\overline{\gamma}^*$ | | 1.31171 | 1.34874 | 1.35825 | 1.35967 |
| C* | M/Sec. | 3514.72 | 2923.15 | 2600.13 | 2394.31 |
| $C_F$ | | 1.20241 | 1.26099 | 1.27776 | 1.28219 |

Necessary data for calculations is taken from the Tables (41), (42), (43) and (44). Calculated data is tabulated as under.

TABLE 51

Theoretically Calculated Data For Performance Parameters For Hydrogen, Oxygen, Nitrogen Propellant System Chemical Rocket Motor At Different O/F Ratios With Frozen Flow Approach, For $p_c$ = 10.20 Ata.

| O/F RATIO | | 2 | 4 | 6 | 8 |
|---|---|---|---|---|---|
| $T_c$ | °K | 3606.40 | 3835.70 | 3902.40 | 3909.40 |
| $T_e$ | °K | 2060.00 | 2230.00 | 2240.00 | 2240.00 |
| $\overline{W}^*$ | Gms/Mole | 5.3005 | 7.9645 | 10.1730 | 12.0130 |
| $v_e$ | M/Sec. | 4611.94 | 3863.10 | 3490.50 | 3325.95 |
| $u_{eq}$ | M/Sec. | 4611.94 | 3863.10 | 3490.50 | 3325.95 |
| $I_{sp}$ | Kgf Sec./Kgm | 470.127 | 393.792 | 355.810 | 328.843 |
| $\overline{\gamma}^*$ | | 1.31775 | 1.30468 | 1.31411 | 1.31544 |
| C* | M/Sec. | 3546.64 | 2994.34 | 2665.65 | 2454.35 |
| $C_F$ | | 1.30037 | 1.29013 | 1.30943 | 1.31438 |

Necessary data for calculations is taken from the Tables (45), (46), (47) and (48). Calculated data is tabulated as under.

TABLE 52

Theoretically Calculated Data For Performance Parameters For Hydrogen, Oxygen, Nitrogen Propellant System Chemical Rocket Motor At Different O/F Ratios With Frozen Flow Approach, For $p_c$ = 13.60 Ata.

| O/F RATIO | | 2 | 4 | 6 | 8 |
|---|---|---|---|---|---|
| $T_c$ | °K | 3686.80 | 3927.40 | 3954.20 | 4017.50 |
| $T_e$ | °K | 2010.00 | 2070.00 | 2200.00 | 2220.00 |
| $\overline{W}^*$ | Gms/Mole | 5.3096 | 7.9780 | 10.1570 | 12.0590 |
| $v_e$ | M/Sec. | 4802.73 | 4149.20 | 3596.30 | 3349.23 |
| $u_{eq}$ | M/Sec. | 4802.73 | 4149.20 | 3596.30 | 3349.23 |
| $I_{sp}$ | Kgf Sec./Kgm | 489.575 | 422.956 | 366.596 | 341.409 |
| $\overline{\gamma}^*$ | | 1.30279 | 1.32515 | 1.29133 | 1.29409 |
| C* | M/Sec. | 3597.26 | 3010.86 | 2705.32 | 2497.67 |
| $C_F$ | | 1.33511 | 1.37808 | 1.32934 | 1.34094 |

B.3 Equilibrium Flow Approach

Solution of the problem discussed in Part 2.4, Section II is attempted here. Calculated data of equilibrium combustion temperature and various molefractions mentioned in Tables (33) to (48) are to be included here as data. Necessary thermodynamic tables for species are to be taken from reference (19). Suitable temperature interval of 200 °K is chosen. Special care is to be taken for assumption of initial guess of number of moles for various species. Over and above their sum, which should be less than exit pressure of one atmosphere, they should be so chosen that RHS constants of the set of equations should approximately come to zero. Error limits are suitably modified to reach convergence. Attached herewith is the computer programme developed for the computer system IBM 7044/1401. All calculations are carried out for small $\alpha$ and $p_c$ = Atmospheric Pressure.

```
CIBJOB           MAP
CIBFTC MAIN
C      PERFORMANCE ANALYSIS OF GAS ROCKET MOTOR USING H2-O2 PROPELLANTS
C      WITH EQUILIBRIUM FLOW TECHNIQUE.
C      READINGS OF VARICUS VALUES MENTICNED IN DIMENSICN STATEMENTS ARE
C      FROM THE STANDARC TABLES AVAILABLE.
C      UNITS OF ENTHALPY ARE KCAL/MOLE.
C      UNITS OF SP-HEAT ARE CAL/OK MOLE.
C      UNITS OF ENTROPY ARE CAL/OK MOLE.
       DIMENSION HOT1(24),HOT2(24),HOT3(24),HOT4(24),HCT5(24),HOT6(24),
      1HOT7(24),HOT8(24),HOT9(24),CPT1(24),CPT2(24),CPT3(24),CPT4(24),
      1CPT5(24),CPT6(24),CPT7(24),CPT8(24),CPT9(24),SOTC1(24),SOTC2(24)
       DIMENSION SOTC3(24),SOTC4(24),SOTC5(24),SOTC6(24),SOTC7(24),
      1SOTC8(24),SOTC9(24),HOTE1(24),HOTE2(24),HOTE3(24),HOTE4(24),
      1HOTE5(24),HOTE6(24),HOTE7(24),HOTE8(24),HOTE9(24)
       DIMENSION CPTE1(24),CPTE2(24),CPTE3(24),CPTE4(24),CPTE5(24),
      1CPTE6(24),CPTE7(24),CPTE8(24),CPTE9(24),SOTE1(24),SOTE2(24),
      1SOTE3(24),SOTE4(24),SOTE5(24),SOTE6(24),SOTE7(24),SOTE8(24),
      1SOTE9(24),AKE1(24),HOF(9),HOTRE(9),T(24),ET(24),TC(4),OFR(4)
       DIMENSION W(9),XJ(4,9),AE(10,11),BE(10,1),XE(10),PE(10),XJE(4,9),
      1AVMWTC(4),AVMWTE(4),AMWR(4),AVMWT(4)
       DIMENSION AKE2(24),AKE3(24),AKE4(24),AKE5(24),AKE6(24)
       REAL NE(9),NJE(9),NTE
       READ 4,(T(I),I=1,24)
     4 FORMAT  (8F[illegible].1)
       READ 6,(AKE1(I),I=1,24)
     6 FORMAT (6(E10.4,2X))
       READ 6,(AKE2(I),I=1,24)
       READ 8,(AKE3(I),I=1,24)
     8 FORMAT (6(E11.5,1X))
       READ 8,(AKE4(I),I=1,24)
       READ 8,(AKE5(I),I=1,24)
       READ 10,(AKE6(I),I=1,24)
    10 FORMAT (4(E14.8,1X))
       READ 12,(HOT1(I),I=1,24)
    12 FORMAT (8F10.4)
       READ 12,(HOT2(I),I=1,24)
       READ 12,(HOT3(I),I=1,24)
       READ 12,(HOT4(I),I=1,24)
       READ 12,(HOT5(I),I=1,24)
       READ 12,(HOT6(I),I=1,24)
       READ 12,(HOT7(I),I=1,24)
       READ 12,(HOT8(I),I=1,24)
       READ 12,(HOT9(I),I=1,24)
       READ 14,(CPT1(I),I=1,24)
    14 FORMAT (8F10.3)
       READ 14,(CPT2(I),I=1,24)
       READ 14,(CPT3(I),I=1,24)
       READ 16,(CPT4(I),I=1,24)
    16 FORMAT (8F10.4)
```

```
      READ 14,(CPT5(I),I=1,24)
      READ 16,(CPT6(I),I=1,24)
      READ 14,(CPT7(I),I=1,24)
      READ 16,(CPT8(I),I=1,24)
      READ 16,(CPT9(I),I=1,24)
      READ 18,(OFR(JJ),JJ=1,4)
   18 FORMAT (4F10.1)
      READ 20,(SOTC1(I),I=1,24)
   20 FORMAT (8F10.4)
      READ 20,(SOTC2(I),I=1,24)
      READ 20,(SOTC3(I),I=1,24)
      READ 20,(SOTC4(I),I=1,24)
      READ 20,(SOTC5(I),I=1,24)
      READ 20,(SOTC6(I),I=1,24)
      READ 20,(SOTC7(I),I=1,24)
      READ 20,(SOTC8(I),I=1,24)
      READ 20,(SOTC9(I),I=1,24)
      READ 4,(ET(I),I=1,24)
      READ 12,(HOTE1(I),I=1,24)
      READ 12,(HOTE2(I),I=1,24)
      READ 12,(HOTE3(I),I=1,24)
      READ 12,(HOTE4(I),I=1,24)
      READ 12,(HOTE5(I),I=1,24)
      READ 12,(HOTE6(I),I=1,24)
      READ 12,(HOTE7(I),I=1,24)
      READ 12,(HOTE8(I),I=1,24)
      READ 12,(HOTE9(I),I=1,24)
      READ 14,(CPTE1(I),I=1,24)
      READ 14,(CPTE2(I),I=1,24)
      READ 14,(CPTE3(I),I=1,24)
      READ 16,(CPTE4(I),I=1,24)
      READ 14,(CPTE5(I),I=1,24)
      READ 16,(CPTE6(I),I=1,24)
      READ 14,(CPTE7(I),I=1,24)
      READ 16,(CPTE8(I),I=1,24)
      READ 16,(CPTE9(I),I=1,24)
      READ 20,(SOTE1(I),I=1,24)
      READ 20,(SOTE2(I),I=1,24)
      READ 20,(SOTE3(I),I=1,24)
      READ 20,(SOTE4(I),I=1,24)
      READ 20,(SOTE5(I),I=1,24)
      READ 20,(SOTE6(I),I=1,24)
      READ 20,(SOTE7(I),I=1,24)
      READ 20,(SOTE8(I),I=1,24)
      READ 20,(SOTE9(I),I=1,24)
      READ 22,(TC(I),I=1,4)
   22 FORMAT (4F10.1)
      READ 24,((XJ(I,II),II=1,9),I=1,4)
   24 FORMAT (3(E10.4,2X))
      READ 26,(HOF(I),I=1,9)
```

```
   26 FORMAT (3F10.4)
      READ 28,(HOTRE(I),I=1,9)
   28 FORMAT (9F8.4)
      READ 30,(W(I),I=1,9)
   30 FORMAT (9F6.1)
      CPTRA=6.80
      EXIPR=1.00
      UGC=1.9857
C     THIS FINISHES THE READING OF ALL NECESSARY DATA.
C     CALCULATION FOR EXIT TEMPARATURE AND THEN FOR PERFORMANCE
C     PARAMETERS STARTS HERE.
C     TO START WITH WE WILL FIRST ASSUME THE EXIT TEMPARATURE AND THEN
C     DETERMINE THE MOLE FRACTIONS OF VARIOUS SPECIES AT THAT EXIT
C     TEMPARATURE AND EXIT PRESSURE OF 1 ATA.
      DO 1111 JJ=1,4
      CALL FLUN(10000)
      ATE=2500.00
  126 J=((ATE-400.00)/200.00)+1.00
      AAKE1=AKE1(J)+((ATE-ET(J))/(ET(J+1)-ET(J)))*(AKE1(J+1)-AKE1(J))
      AAKE2=AKE2(J)+((ATE-ET(J))/(ET(J+1)-ET(J)))*(AKE2(J+1)-AKE2(J))
      AAKE3=AKE3(J)+((ATE-ET(J))/(ET(J+1)-ET(J)))*(AKE3(J+1)-AKE3(J))
      AAKE4=AKE4(J)+((ATE-ET(J))/(ET(J+1)-ET(J)))*(AKE4(J+1)-AKE4(J))
      AAKE5=AKE5(J)+((ATE-ET(J))/(ET(J+1)-ET(J)))*(AKE5(J+1)-AKE5(J))
      AAKE6=AKE6(J)+((ATE-ET(J))/(ET(J+1)-ET(J)))*(AKE6(J+1)-AKE6(J))
      KKK=J
C      INITIAL GUESS FOR VARIOUS VALUES OF NO OF MOLES.
      NE(1)=0.50
      NE(2)=0.10
      NE(3)=0.004
      NE(4)=0.10
      NE(5)=0.10
      NE(6)=0.01
      NE(7)=0.05
      NE(8)=0.05
      NE(9)=0.0001
      ANEQWE=1.00
  128 SUMNE=NE(1)+NE(2)+NE(3)+NE(4)+NE(5)+NE(6)+NE(7)+NE(8)+NE(9)
      ARE1TR=AAKE1
      ARE2TR=AAKE2
      ARE3TR=AAKE3
      ARE4TR=AAKE4
      ARE5TR=AAKE5
      ARE6TR=AAKE6
      ATRUEE=(16.00/OFR(JJ))*2.00
      BTRUEE=2.00
      FACOIM=((16.00/OFR(JJ))+1.00)*0.001
      CTRUEE=FACOIM*2.00
      AACALE=2.00*NE(1)+NE(2)+2.00*NE(4)+NE(7)
      BACALE=NE(1)+NE(2)+NE(3)+2.00*NE(5)+NE(8)
      CACALE=NE(3)+2.00*NE(6)+NE(9)
```

```
      ACALE=AACALE/ANEQWE
      BCALE=BACALE/ANEQWE
      CCALE=CACALE/ANEQWE
      PTRAE=EXIPR
      PCALE=SUMNE
      ARE1CA=(NE(1))/(NE(4)*(NE(5)**0.50))
      ARE2CA=((NE(2))*(NE(4)**0.50))/(NE(1))
      ARE3CA=(NE(3))/((NE(6)**0.50)*(NE(5)**0.50))
      ARE4CA=(NE(7))/(NE(4)**0.50)
      ARE5CA=(NE(8))/(NE(5)**0.50)
      ARE6CA=(NE(9))/(NE(6)**0.50)
C     PREREQUISITE CALCULATIONS END HERE.
C     CALCULATION FOR ELEMENTS OF MATRIX STARTS HERE.
      DO 70 II=1,10
      DO 70 KK=1,11
      AE(II,KK)=0.00
   70 CONTINUE
      AE(1,1)=NE(1)
      AE(1,2)=NE(2)
      AE(1,3)=NE(3)
      AE(1,4)=NE(4)
      AE(1,5)=NE(5)
      AE(1,6)=NE(6)
      AE(1,7)=NE(7)
      AE(1,8)=NE(8)
      AE(1,9)=NE(9)
      AE(1,11)=PCALE*ALOG(PTRAE/PCALE)
      AE(2,1)=NE(1)
      AE(2,2)=NE(2)
      AE(2,3)=NE(3)
      AE(2,5)=2.00*NE(5)
      AE(2,8)=NE(8)
      AE(2,10)=-(BACALE)
      AE(2,11)=(BACALE)*ALOG(BTRUEE/BCALE)
      AE(3,3)=NE(3)
      AE(3,6)=2.00*NE(6)
      AE(3,9)=NE(9)
      AE(3,10)=-(CACALE)
      AE(3,11)=(CACALE)*ALOG(CTRUEE/CCALE)
      AE(4,1)=-1.00
      AE(4,2)=1.00
      AE(4,4)=+0.50
      AE(4,11)=ALOG(ARE2TR/ARE2CA)
      AE(5,1)=1.00
      AE(5,4)=-1.00
      AE(5,5)=-0.50
      AE(5,11)=ALOG(ARE1TR/ARE1CA)
      AE(6,3)=1.00
      AE(6,5)=-0.50
      AE(6,6)=-0.50
```

```
      AE(6,11)=ALOG(ARE3TR/ARE3CA)
      AE(7,4)=-0.50
      AE(7,7)=1.00
      AE(7,11)=ALOG(ARE4TR/ARE4CA)
      AE(8,5)=-0.50
      AE(8,8)=1.00
      AE(8,11)=ALOG(ARE5TR/ARE5CA)
      AE(9,6)=-0.50
      AE(9,9)=1.00
      AE(9,11)=ALOG(ARE6TR/ARE6CA)
      AE(10,1)=2.00*NE(1)
      AE(10,2)=NE(2)
      AE(10,4)=2.00*NE(4)
      AE(10,7)=NE(7)
      AE(10,10)=-(AACALE)
      AE(10,11)=(AACALE)*ALOG(ATRUEE/ACALE)
      DO 130 I=1,10
  130 BE(I,1)=AE(I,11)
      CALL MATINV(AE,10,BE,1,DET)
      DO 144 J=1,10
  144 XE(J)=BE(J,1)
      DO 149 I=1,9
      PE(I)=XE(I)+ALOG(NE(I))
      NE(I)=EXP(PE(I))
  149 CONTINUE
      PE(10)=XE(10)+ALOG(ANEQWE)
      ANEQWE=EXP(PE(10))
C     HERE STARTS THE CHECKING FOR PP ADJUSTMENT.
      SNEPPA=NE(1)+NE(2)+NE(3)+NE(4)+NE(5)+NE(6)+NE(7)+NE(8)+NE(9)
      PERROE=((SNEPPA-EXIPR)/EXIPR)*100.00
      IF(ABS(PERROE).LE.1.00) GO TO 162
      GO TO 128
  162 NTE=SNEPPA/ANEQWE
      DO 166 I=1,9
      NJE(I)=NE(I)/ANEQWE
      XJE(JJ,I)=NJE(I)/NTE
  166 CONTINUE
      AVMWTC(JJ)=XJ(JJ,1)*W(1)+XJ(JJ,2)*W(2)+XJ(JJ,3)*W(3)+XJ(JJ,4)*W(4)
     1+XJ(JJ,5)*W(5)+XJ(JJ,6)*W(6)+XJ(JJ,7)*W(7)+XJ(JJ,8)*W(8)+XJ(JJ,9)*
     1W(9)
      AVMWTE(JJ)=XJE(JJ,1)*W(1)+XJE(JJ,2)*W(2)+XJE(JJ,3)*W(3)+XJE(JJ,4)*
     1W(4)+XJE(JJ,5)*W(5)+XJE(JJ,6)*W(6)+XJE(JJ,7)*W(7)+XJE(JJ,8)*W(8)+
     1XJE(JJ,9)*W(9)
      AVMWT(JJ)=(AVMWTC(JJ)+AVMWTE(JJ))/2.00
      AMWR(JJ)=AVMWTC(JJ)/AVMWTE(JJ)
C     INTERPOLATION FOR ENTROPY STARTS HERE.
      ATC=TC(JJ)
      I=((ATC-400.00)/200.00)+1.00
      ASOTC1=SOTC1(I)+((ATC-T(I))/(T(I+1)-T(I)))*(SOTC1(I+1)-SOTC1(I))
      ASOTC2=SOTC2(I)+((ATC-T(I))/(T(I+1)-T(I)))*(SOTC2(I+1)-SOTC2(I))
```

```
      ASOTC3=SOTC3(I)+((ATC-T(I))/(T(I+1)-T(I)))*(SOTC3(I+1)-SOTC3(I))
      ASOTC4=SOTC4(I)+((ATC-T(I))/(T(I+1)-T(I)))*(SOTC4(I+1)-SOTC4(I))
      ASOTC5=SOTC5(I)+((ATC-T(I))/(T(I+1)-T(I)))*(SOTC5(I+1)-SOTC5(I))
      ASOTC6=SOTC6(I)+((ATC-T(I))/(T(I+1)-T(I)))*(SOTC6(I+1)-SOTC6(I))
      ASOTC7=SOTC7(I)+((ATC-T(I))/(T(I+1)-T(I)))*(SOTC7(I+1)-SOTC7(I))
      ASOTC8=SOTC8(I)+((ATC-T(I))/(T(I+1)-T(I)))*(SOTC8(I+1)-SOTC8(I))
      ASOTC9=SOTC9(I)+((ATC-T(I))/(T(I+1)-T(I)))*(SOTC9(I+1)-SOTC9(I))
      J=KKK
      ASOTE1=SOTE1(J)+((ATE-ET(J))/(ET(J+1)-ET(J)))*(SOTE1(J+1)-SOTE1(J)
     1)
      ASOTE2=SOTE2(J)+((ATE-ET(J))/(ET(J+1)-ET(J)))*(SOTE2(J+1)-SOTE2(J)
     1)
      ASOTE3=SOTE3(J)+((ATE-ET(J))/(ET(J+1)-ET(J)))*(SOTE3(J+1)-SOTE3(J)
     1)
      ASOTE4=SOTE4(J)+((ATE-ET(J))/(ET(J+1)-ET(J)))*(SOTE4(J+1)-SOTE4(J)
     1)
      ASOTE5=SOTE5(J)+((ATE-ET(J))/(ET(J+1)-ET(J)))*(SOTE5(J+1)-SOTE5(J)
     1)
      ASOTE6=SOTE6(J)+((ATE-ET(J))/(ET(J+1)-ET(J)))*(SOTE6(J+1)-SOTE6(J)
     1)
      ASOTE7=SOTE7(J)+((ATE-ET(J))/(ET(J+1)-ET(J)))*(SOTE7(J+1)-SOTE7(J)
     1)
      ASOTE8=SOTE8(J)+((ATE-ET(J))/(ET(J+1)-ET(J)))*(SOTE8(J+1)-SOTE8(J)
     1)
      ASOTE9=SOTE9(J)+((ATE-ET(J))/(ET(J+1)-ET(J)))*(SOTE9(J+1)-SOTE9(J)
     1)
C      CHECK FOR EXIT TEMPARATURE STARTS HERE.
      RHSE=UGC*ALOG(CPTRA)
      TERS1=XJ(JJ,1)*(ASOTC1-UGC*ALOG(XJ(JJ,1)))-AMWR(JJ)*XJE(JJ,1)*
     1(ASOTE1-UGC*ALOG(XJE(JJ,1)))
      TERS2=XJ(JJ,2)*(ASOTC2-UGC*ALOG(XJ(JJ,2)))-AMWR(JJ)*XJE(JJ,2)*
     1(ASOTE2-UGC*ALOG(XJE(JJ,2)))
      TERS3=XJ(JJ,3)*(ASOTC3-UGC*ALOG(XJ(JJ,3)))-AMWR(JJ)*XJE(JJ,3)*
     1(ASOTE3-UGC*ALOG(XJE(JJ,3)))
      TERS4=XJ(JJ,4)*(ASOTC4-UGC*ALOG(XJ(JJ,4)))-AMWR(JJ)*XJE(JJ,4)*
     1(ASOTE4-UGC*ALOG(XJE(JJ,4)))
      TERS5=XJ(JJ,5)*(ASOTC5-UGC*ALOG(XJ(JJ,5)))-AMWR(JJ)*XJE(JJ,5)*
     1(ASOTE5-UGC*ALOG(XJE(JJ,5)))
      TERS6=XJ(JJ,6)*(ASOTC6-UGC*ALOG(XJ(JJ,6)))-AMWR(JJ)*XJE(JJ,6)*
     1(ASOTE6-UGC*ALOG(XJE(JJ,6)))
      TERS7=XJ(JJ,7)*(ASOTC7-UGC*ALOG(XJ(JJ,7)))-AMWR(JJ)*XJE(JJ,7)*
     1(ASOTE7-UGC*ALOG(XJE(JJ,7)))
      TERS8=XJ(JJ,8)*(ASOTC8-UGC*ALOG(XJ(JJ,8)))-AMWR(JJ)*XJE(JJ,8)*
     1(ASOTE8-UGC*ALOG(XJE(JJ,8)))
      TERS9=XJ(JJ,9)*(ASOTC9-UGC*ALOG(XJ(JJ,9)))-AMWR(JJ)*XJE(JJ,9)*
     1(ASOTE9-UGC*ALOG(XJE(JJ,9)))
      SUMSE=TERS1+TERS2+TERS3+TERS4+TERS5+TERS6+TERS7+TERS8+TERS9
      PERR=((SUMSE-RHSE)/RHSE)*100.00
      PRINT 299
  299 FORMAT (* PERR      ATE *)
```

```
      PRINT 300,PERR,ATE
  300 FORMAT (1X,2E14.5)
      IF (ABS(PERR).GE.60.00) GO TO 195
      IF (ABS(PERR).LE.3.00) GO TO 175
      ATE=ATE+10.00
      GO TO 126
  195 ATE=ATE+100.00
      GO TO 126
  175 PRINT 176
  176 FORMAT (//,10X,* RHSE      SUMSE *)
      PRINT 177,RHSE,SUMSE
  177 FORMAT (1X,2E14.5)
C     CALCULATION FOR EXIT TEMPARATURE DETERMINATION ENDS HERE.
C     CALCULATION FOR PERFORMANCE PARAMETERS STARTS HERE.
C     INTERPOLATIONS NECESSARY ARE STARTED.
      ACPT1=CPT1(I)+((ATC-T(I))/(T(I+1)-T(I)))*(CPT1(I+1)-CPT1(I))
      ACPT2=CPT2(I)+((ATC-T(I))/(T(I+1)-T(I)))*(CPT2(I+1)-CPT2(I))
      ACPT3=CPT3(I)+((ATC-T(I))/(T(I+1)-T(I)))*(CPT3(I+1)-CPT3(I))
      ACPT4=CPT4(I)+((ATC-T(I))/(T(I+1)-T(I)))*(CPT4(I+1)-CPT4(I))
      ACPT5=CPT5(I)+((ATC-T(I))/(T(I+1)-T(I)))*(CPT5(I+1)-CPT5(I))
      ACPT6=CPT6(I)+((ATC-T(I))/(T(I+1)-T(I)))*(CPT6(I+1)-CPT6(I))
      ACPT7=CPT7(I)+((ATC-T(I))/(T(I+1)-T(I)))*(CPT7(I+1)-CPT7(I))
      ACPT8=CPT8(I)+((ATC-T(I))/(T(I+1)-T(I)))*(CPT8(I+1)-CPT8(I))
      ACPT9=CPT9(I)+((ATC-T(I))/(T(I+1)-T(I)))*(CPT9(I+1)-CPT9(I))
      CPT1B=(ACPT1+CPT1(I))/2.00
      CPT2B=(ACPT2+CPT2(I))/2.00
      CPT3B=(ACPT3+CPT3(I))/2.00
      CPT4B=(ACPT4+CPT4(I))/2.00
      CPT5B=(ACPT5+CPT5(I))/2.00
      CPT6B=(ACPT6+CPT6(I))/2.00
      CPT7B=(ACPT7+CPT7(I))/2.00
      CPT8B=(ACPT8+CPT8(I))/2.00
      CPT9B=(ACPT9+CPT9(I))/2.00
      AHOT1=HOT1(I)+((CPT1B*(ATC-T(I)))/1000.00)
      AHOT2=HOT2(I)+((CPT2B*(ATC-T(I)))/1000.00)
      AHOT3=HOT3(I)+((CPT3B*(ATC-T(I)))/1000.00)
      AHOT4=HOT4(I)+((CPT4B*(ATC-T(I)))/1000.00)
      AHOT5=HOT5(I)+((CPT5B*(ATC-T(I)))/1000.00)
      AHOT6=HOT6(I)+((CPT6B*(ATC-T(I)))/1000.00)
      AHOT7=HOT7(I)+((CPT7B*(ATC-T(I)))/1000.00)
      AHOT8=HOT8(I)+((CPT8B*(ATC-T(I)))/1000.00)
      AHOT9=HOT9(I)+((CPT9B*(ATC-T(I)))/1000.00)
      ACPTE1=CPTE1(J)+((ATE-ET(J))/(ET(J+1)-ET(J)))*(CPTE1(J+1)-CPTE1(J)
     1)
      ACPTE2=CPTE2(J)+((ATE-ET(J))/(ET(J+1)-ET(J)))*(CPTE2(J+1)-CPTE2(J)
     1)
      ACPTE3=CPTE3(J)+((ATE-ET(J))/(ET(J+1)-ET(J)))*(CPTE3(J+1)-CPTE3(J)
     1)
      ACPTE4=CPTE4(J)+((ATE-ET(J))/(ET(J+1)-ET(J)))*(CPTE4(J+1)-CPTE4(J)
     1)
```

```
      ACPTE5=CPTE5(J)+((ATE-ET(J))/(ET(J+1)-ET(J)))*(CPTE5(J+1)-CPTE5(J)
     1)
      ACPTE6=CPTE6(J)+((ATE-ET(J))/(ET(J+1)-ET(J)))*(CPTE6(J+1)-CPTE6(J)
     1)
      ACPTE7=CPTE7(J)+((ATE-ET(J))/(ET(J+1)-ET(J)))*(CPTE7(J+1)-CPTE7(J)
     1)
      ACPTE8=CPTE8(J)+((ATE-ET(J))/(ET(J+1)-ET(J)))*(CPTE8(J+1)-CPTE8(J)
     1)
      ACPTE9=CPTE9(J)+((ATE-ET(J))/(ET(J+1)-ET(J)))*(CPTE9(J+1)-CPTE9(J)
     1)
      CPTE1B=(ACPTE1+CPTE1(J))/2.00
      CPTE2B=(ACPTE2+CPTE2(J))/2.00
      CPTE3B=(ACPTE3+CPTE3(J))/2.00
      CPTE4B=(ACPTE4+CPTE4(J))/2.00
      CPTE5B=(ACPTE5+CPTE5(J))/2.00
      CPTE6B=(ACPTE6+CPTE6(J))/2.00
      CPTE7B=(ACPTE7+CPTE7(J))/2.00
      CPTE8B=(ACPTE8+CPTE8(J))/2.00
      CPTE9B=(ACPTE9+CPTE9(J))/2.00
      AHOTE1=HOTE1(J)+((CPTE1B*(ATE-ET(J)))/1000.00)
      AHOTE2=HOTE2(J)+((CPTE2B*(ATE-ET(J)))/1000.00)
      AHOTE3=HOTE3(J)+((CPTE3B*(ATE-ET(J)))/1000.00)
      AHOTE4=HOTE4(J)+((CPTE4B*(ATE-ET(J)))/1000.00)
      AHOTE5=HOTE5(J)+((CPTE5B*(ATE-ET(J)))/1000.00)
      AHOTE6=HOTE6(J)+((CPTE6B*(ATE-ET(J)))/1000.00)
      AHOTE7=HOTE7(J)+((CPTE7B*(ATE-ET(J)))/1000.00)
      AHOTE8=HOTE8(J)+((CPTE8B*(ATE-ET(J)))/1000.00)
      AHOTE9=HOTE9(J)+((CPTE9B*(ATE-ET(J)))/1000.00)
      TERH1=XJ(JJ,1)*(HOF(1)+AHOT1-HOTRE(1))-AMWR(JJ)*XJE(JJ,1)*(HOF(1)+
     1AHOTE1-HOTRE(1))
      TERH2=XJ(JJ,2)*(HOF(2)+AHOT2-HOTRE(2))-AMWR(JJ)*XJE(JJ,2)*(HOF(2)+
     1AHOTE2-HOTRE(2))
      TERH3=XJ(JJ,3)*(HOF(3)+AHOT3-HOTRE(3))-AMWR(JJ)*XJE(JJ,3)*(HOF(3)+
     1AHOTE3-HOTRE(3))
      TERH4=XJ(JJ,4)*(HOF(4)+AHOT4-HOTRE(4))-AMWR(JJ)*XJE(JJ,4)*(HOF(4)+
     1AHOTE4-HOTRE(4))
      TERH5=XJ(JJ,5)*(HOF(5)+AHOT5-HOTRE(5))-AMWR(JJ)*XJE(JJ,5)*(HOF(5)+
     1AHOTE5-HOTRE(5))
      TERH6=XJ(JJ,6)*(HOF(6)+AHOT6-HOTRE(6))-AMWR(JJ)*XJE(JJ,6)*(HOF(6)+
     1AHOTE6-HOTRE(6))
      TERH7=XJ(JJ,7)*(HOF(7)+AHOT7-HOTRE(7))-AMWR(JJ)*XJE(JJ,7)*(HOF(7)+
     1AHOTE7-HOTRE(7))
      TERH8=XJ(JJ,8)*(HOF(8)+AHOT8-HOTRE(8))-AMWR(JJ)*XJE(JJ,8)*(HOF(8)+
     1AHOTE8-HOTRE(8))
      TERH9=XJ(JJ,9)*(HOF(9)+AHOT9-HOTRE(9))-AMWR(JJ)*XJE(JJ,9)*(HOF(9)+
     1AHOTE9-HOTRE(9))
      SUMHE=TERH1+TERH2+TERH3+TERH4+TERH5+TERH6+TERH7+TERH8+TERH9
      PERFORMANCE CALCULATION FOR SMALL ALPHA AND PEXIT=PATMA STARTS.
      MKS UNITS ARE USED HERE.
      VEXITE=((SQRT(SUMHE))*(SQRT(8370.0))/(SQRT(AVMWT(JJ))))*(SQRT(
```

```
     11000.00))
      EVEXIE=VEXITE
      SPIMPE=VEXITE/9.81
      GSTARE=(ALOG(CPTRA))/(ALOG(CPTRA)-ALOG((ATC/ATE)))
      CGSTAE=(2.0/(GSTARE+1.0))**((GSTARE+1.0)/(2.0*(GSTARE-1.0)))
      CHVELE=(1.0/CGSTAE)*SQRT((UGC*ATC)/(AVMWT(JJ)*GSTARE))*SQRT(4185.0
     1)
      THCOFE=EVEXIE/CHVELE
      PRINT 200
  200 FORMAT (* COMBUSTION CHAMBER PRESSURE      OFR *)
      PRINT 201,CPTRA,OFR(JJ)
  201 FORMAT (1X,2E14.5)
      PRINT 202
  202 FORMAT (* COMBUSTION TEMPARATURE      EXIT TEMPARATURE *)
      PRINT 203,TC(JJ),ATE
  203 FORMAT (1X,2E14.5)
      PRINT 400
  400 FORMAT (* MOLE FRACTION OF VARIOUS SPECIES AT EXIT TEMPARATURE *)
      PRINT 401,(XJE(JJ,I),I=1,9)
  401 FORMAT (1X,9E14.5)
      PRINT 204
  204 FORMAT (* AVMWTC    AVMWTE    AMWR    AVMWT *)
      PRINT 205,AVMWTC(JJ),AVMWTE(JJ),AMWR(JJ),AVMWT(JJ)
  205 FORMAT (1X,4E14.5)
      PRINT 206
  206 FORMAT (* VEXITE EVEXITE SPIMPE GSTARE CHVELE THCOFE *)
      PRINT 207,VEXITE,EVEXIE,SPIMPE,GSTARE,CHVELE,THCOFE
  207 FORMAT (1X,6E14.5)
 1111 CONTINUE
      STOP
      END
CFETCH MATINV  CCS999
      SUBROUTINE MATINV(C,N,B,M,DETERM)                                 MAT0002(
      DIMENSION C(10,11),B(10,1),IPIVOT(10),INDEX(10,2)                 MAT0011(
      DOUBLE PRECISION A(10,10),AMAX,T,SWAP,PIVOT                       MAT0011
      DO 5 I=1,N                                                        MAT0013
      DO 5 J=1,N                                                        MAT0013
    5 A(I,J)=C(I,J)                                                     MAT0013
   80 IF(AMAX-DABS(A(J,K)))85,100,100                                   MAT0024
   95 AMAX=DABS(A(J,K))                                                 MAT0027
  320 DETERM=DETERM*PIVOT/DABS(PIVOT)                                   MAT0047
      DO 780 I=1,N                                                      MAT0077
      DO 780 J=1,N                                                      MAT0077
  780 C(I,J)=A(I,J)                                                     MAT0077
CDONE
CENTRY
C     NOW FOLLOWS THE NECESSARY DATA CARDS.
```

Necessary data for calculations is taken from the Tables (33), (34), (35) and (36). Calculated data is tabulated as under.

TABLE 53

Theoretically Calculated Data For Performance Parameters For Hydrogen, Oxygen, Nitrogen Propellant System Chemical Rocket Motor At Different O/F Ratios, With Equilibrium Flow Approach, $p_c$ = 3.40 Ata.

| O/F RATIO | | 2 | 4 | 6 | 8 |
|---|---|---|---|---|---|
| $T_c$ | °K | 3393.00 | 3607.40 | 3629.90 | 3674.50 |
| $T_e$ | °K | 3030.00 | 3260.00 | 3340.00 | 3340.00 |
| $\bar{W}^*$ | Gms/Mole | 5.4204 | 8.1573 | 10.3730 | 12.3230 |
| $v_e$ | M/Sec | 3505.10 | 2921.80 | 2484.20 | 2371.10 |
| $u_{eq}$ | M/Sec | 3505.10 | 2921.80 | 2484.20 | 2371.10 |
| $I_{sp}$ | Kgf Sec/Kgm | 357.30 | 297.84 | 253.24 | 241.70 |
| $\bar{\gamma}^*$ | | 1.1019 | 1.0902 | 1.0730 | 1.0846 |
| C* | M/Sec. | 3627.40 | 3060.90 | 2738.90 | 2518.20 |
| $C_F$ | | 0.96628 | 0.95455 | 0.90704 | 0.94160 |

Necessary data for calculations is taken from the tables (37), (38), (39) and (40). Calculated data is tabulated as under.

TABLE 54

Theoretically Calculated Data For Performance Parameters For Hydrogen, Oxygen, Nitrogen Propellant System Chemical Rocket Motor At Different O/F Ratios, With Equilibrium Flow Approach, $p_c$ = 6.80 Ata.

| O/F RATIO | 2 | 4 | 6 | 8 |
|---|---|---|---|---|
| $T_c$ °K | 3532.50 | 3742.80 | 3796.80 | 3802.40 |
| $T_e$ °K | 2950.00 | 3220.00 | 3290.00 | 3310.00 |
| $\bar{W}$ * Gms/Mole | 5.4722 | 8.2447 | 10.5510 | 12.4310 |
| $v_e$ M/Sec. | 4268.30 | 3576.90 | 3227.80 | 2922.10 |
| $u_{eq}$ M/Sec. | 4268.30 | 3576.90 | 3227.80 | 2922.10 |
| $I_{sp}$ Kgf Sec./Kgm | 435.09 | 364.61 | 329.03 | 297.87 |
| $\bar{\gamma}$ * | 1.1038 | 1.0852 | 1.0808 | 1.0780 |
| C* M/Sec. | 3681.40 | 3106.50 | 2769.90 | 2556.30 |
| $C_F$ | 1.1594 | 1.1514 | 1.1653 | 1.1431 |

Necessary data for calculations is taken from the Tables (41), (42), (43) and (44). Calculated data is tabulated as under.

TABLE 55

Theoretically Calculated Data For Performance Parameters For Hydrogen, Oxygen, Nitrogen Propellant System Chemical Rocket Motor At Different O/F Ratios, With Equilibrium Flow Approach, $p_c$ = 10.20 Ata.

| O/F RATIO | | 2 | 4 | 6 | 8 |
|---|---|---|---|---|---|
| $T_c$ | °K | 3606.40 | 3835.70 | 3902.40 | 3909.40 |
| $T_e$ | °K | 2880.00 | 3190.00 | 3270.00 | 3280.00 |
| $\bar{W}^*$ | Gms/Mole | 5.5050 | 8.3076 | 10.6230 | 12.5720 |
| $v_e$ | M/Sec. | 4732.20 | 3944.20 | 3518.20 | 3281.90 |
| $u_{eq}$ | M/Sec. | 4732.20 | 3944.20 | 3518.20 | 3281.90 |
| $I_{sp}$ | Kgf Sec./ Kgm | 482.39 | 402.06 | 358.63 | 334.55 |
| $\bar{\gamma}^*$ | | 1.1072 | 1.0862 | 1.0824 | 1.0818 |
| C* | M/Sec. | 3704.30 | 3131.00 | 2797.10 | 2574.00 |
| $C_F$ | | 1.2775 | 1.2594 | 1.2578 | 1.2750 |

Necessary data for calculations is taken from the Tables (45), (46), (47) and (48). Calculated data is tabulated as under.

TABLE 56

Theoretically Calculated Data For Performance Parameters For Hydrogen, Oxygen, Nitrogen Propellant System Chemical Rocket Motor At Different O/F Ratios, With Equilibrium Flow Approach, $p_c$ = 15.60 Ata

| O/F RATIO | | 2 | 4 | 6 | 8 |
|---|---|---|---|---|---|
| $T_c$ | °K | 3686.80 | 3927.40 | 3954.20 | 4017.50 |
| $T_e$ | °K | 2840.00 | 3160.00 | 3250.00 | 3270.00 |
| $\overline{W}$ * | Gms/Mole | 5.5291 | 8.3655 | 10.6850 | 12.6400 |
| $v_e$ | M/Sec. | 4979.20 | 4203.70 | 3758.80 | 3419.40 |
| $u_{eq}$ | M/Sec. | 4979.20 | 4203.70 | 3758.80 | 3419.40 |
| $I_{sp}$ | Kgf Sec./Kgm | 507.56 | 428.51 | 383.16 | 348.56 |
| $\bar{\gamma}$ * | | 1.1111 | 1.0909 | 1.0824 | 1.0856 |
| C* | M/Sec. | 3732.50 | 3153.10 | 2811.10 | 2599.00 |
| $C_F$ | | 1.3340 | 1.3332 | 1.3372 | 1.3157 |